NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

TECHNICAL NOTE

No. 1349

PERFORMANCE AND RANGES OF APPLICATION OF VARIOUS TYPES

OF AIRCRAFT-PROPULSION SYSTEM

By Cleveland Laboratory Staff

Flight Propulsion Research Laboratory
Cleveland, Ohio

Washington
August 1947

PREFACE

This group of papers on the comparison of the performance
of six aircraft-propulsion systems was prepared by members of the
NACA Flight Propulsion Research Laboratory staff under the direc-
tion of Mr. Benjamin Pinkel and was presented at the meeting of
the Institute of Aeronautical Sciences on Aircraft Propulsion
Systems held in Cleveland, Ohio, on March 28, 1947.

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SUMMARY

A discussion of the performance characteristics of (1) the
compound engine, (2) the turbine-propeller engine, (3) the
turbojet engine, (4) the turbo-ram-jet engine, (5) the ram-jet
engine, and (6) the rocket engine is presented. An insight is
provided into the proper position of each of these engine types
in the speed-range spectrum of aircraft operation. Both subsonic
and supersonic flight are considered.

It is shown that the compound engine, which has the greatest
weight per unit thrust and also the lowest specific fuel con-
sumption, gives the longest range. As the speed is increased, the
increased engine weight and nacelle drag result in a reduction in
the disposable load that the airplane is capable of carrying and
hence in a reduction in the range. Therefore, as speed is
increased it is necessary to progress to engine types that provide
greater thrust per unit weight and per unit frontal area, gener-
ally at the cost of an increased specific fuel consumption and
resultant decreased range. It is shown that the turbine-propeller
engine provides better performance on the basis of current values
of weight per unit thrust than the other engines considered at
moderate speeds and altitudes but that a large reduction in weight
per unit thrust is required in this type of engine to make it
suitable for high-speed operation in the subsonic range. At high-
speed flight in the subsonic range it is desirable to shift to the
turbojet engine.

At supersonic speeds, the range of the airplane increases
with increased flight speed and altitude for each of the propulsion
systems considered. The ram jet gives the longest range of the
power plants considered and is approached by the turbo-ram-jet
engine only when it approaches the ram jet in operation, that is,
when the pressure ratio across the compressor reaches unity in
value. The turbo-ram-jet engine, however, has an advantage over
the ram-jet engine in that it can be designed to provide the com-
bination of appreciable thrust for take-off and good high-speed
performance.

The rocket engine when applied to an airplane, because of its low weight per unit thrust and its compactness, gives the highest disposable load, but because of its extremely high specific propellant consumption gives the shortest range.

INTRODUCTION

This group of papers is presented to provide an insight into the most suitable aircraft operational ranges of six types of propulsion system now under development. A comparison is made of the performance characteristics of (1) the compound engine, (2) the turbine-propeller engine, (3) the turbojet engine, (4) the turbo-ram-jet engine, (5) the ram-jet engine, and (6) the rocket engine. The position of each of these engine types in the aircraft speed-range spectrum is indicated.

The position of each of these engine types in the speed-range spectrum of aircraft operation is dependent on the assumptions made with regard to the power plant and the airplane. No single set of assumptions satisfy all types of aircraft application and operational procedure. Improvements in the design of the engine and the associated airplane influence the results. Furthermore, at flight conditions where only a small difference in performance exists between two engine types, the choice of power plant is determined by such factors as simplicity of design and installation, ease of maintenance, cost of the engine, reliability, and availability of the desired size. Therefore, it is not the purpose of these papers to define precisely the zones of flight operation for each engine type but to provide an approximate indication as a basis for illustrating the relation between the engine characteristics and the position of the engine in the flight-operational spectrum.

With this limited objective, no attempt was made to design the best airplane for each engine type nor to lay out the best flight plan, but rather to set up the simple assumptions listed in the appendix with the belief that they are not overly prejudicial to any one of the engine types. Subsonic and supersonic flight-speed ranges and accompanying differences in design and performance characteristics are considered.

Each combination of flight speed and altitude in the analysis is considered a design point in that the engine is assumed to be designed specifically for cruise operation at that point. A complete analysis of any engine for a specific application requires a consideration of the performance of a fixed engine over a range of conditions some of which may be far from the design point. The various engine types differ in the sensitivity of their performance to shift in conditions from the design point. In particular some engines provide greater thrust for take-off and climb than do others for equal thrust in the cruise condition, and some are more adaptable for application of thrust augmentation methods for these

short-duration operations. A complete study that considers these
factors would involve many arbitrary assumptions. The present
analysis was limited therefore to a consideration of a series of
design points at the cruise conditions with the belief that the
results would be indicative provided that the limitations of the
analysis are clearly born in mind.

The weight of the engine per unit thrust is an important
factor in the determination of the load-carrying capacity of an
airplane equipped with the engine. The higher the weight of the
power plant for a given thrust, the lower, of course, is the dis-
posable load that may be carried by the airplane. In the case of
the compound, the turbojet, and the turbine-propeller engines, the
weight estimates were guided by a consideration of the weights of
engines that have been built and tested. The components of the
compound engine, namely, the reciprocating engine, the exhaust-gas
turbine, and the supercharger, have been the subject of intensive
development over a long period of time and no large changes in
weight in conventional designs of these components are anticipated.
The development of the two turbine engines is recent and a signifi-
cant reduction in weight per unit thrust may be achieved by refine-
ment in design, improvement in materials, and increase in permissible
gas temperatures through the use of turbine cooling. On the other
hand, efforts to provide greater life and adequate automatic control
tend to increase engine weight. The comparison of these power plants
on the basis of weight is therefore transitory. The improvement in
performance of an airplane equipped with turbine-propeller engines
that results from a reduction in engine weight is discussed. The
results of the analysis are plotted in a form permitting rapid
evaluation of the improvement in airplane performance that can be
obtained with a reduction in engine weight.

The performance values of the turbine engines presented are
based on component efficiencies that have been achieved in labora-
tory investigations on research compressors and turbines designed
for high efficiencies. These efficiencies have not yet been obtained
on components of current turbine engines. Although the specific
fuel consumptions used in this analysis for the turbine engines are
considerably better than obtained in current practice, they are not
outside reasonable expectation.

The comparison of the actual performance of airplanes equipped
with various types of engine must take into account such factors as
flight plan, part throttle efficiency, reserve fuel for emergency,
division of disposable load between pay load and fuel, and other
practical considerations. These considerations change with type of
application and with time. It was therefore considered undesirable

to sacrifice generality by arbitrary assumptions in this connection.
Instead the performance of the various engines are presented in a
form to illustrate their essential characteristics and to permit
application of any desired assumptions as to flight operation.

At any given flight speed and altitude, the merit of a given
propulsion system is judged on: (a) the percentage of initial
gross weight still available for disposable load (fuel load plus
pay load) after the weight of the propulsion system required to
obtain the desired performance is deducted; and (b) the rate per
mile that disposable load is consumed (as fuel) per ton of initial
airplane gross weight to fly at the desired speed and altitude.
The ratio of (a) to (b) is the approximate maximum range for the
given application.

The results of the computations are summarized by curves for
all of the engine types at various speeds and altitudes plotted
with the disposable load per pound of gross weight as the ordinate,
the fuel rate in pounds per mile per ton of gross weight as the
abscissa, and the approximate range as a third scale. A factor
that corrects the approximate range for the effect of the change
in the gross weight of the airplane during the flight is also
shown. In such a plot, it is expected that for any one engine
type as the flight speed is increased the disposable load per
pound of airplane weight is decreased because of the increased
engine weight required to supply the increased thrust, and that
a speed is reached at which it becomes desirable to shift to an
engine type having a lower weight per unit thrust in order to
restore the disposable load even if it results in an increase in
fuel rate per ton-mile. Thus the trend toward increased speeds
is expected to be accompanied by a shift toward engine types
having lower weight per net thrust usually at the cost of an
increased fuel consumption.

The performance characteristics of the various propulsion
systems and their position in the operational spectrum are dis-
cussed in the individual sections of this report and are sum-
marized in a final section.

I - THE COMPOUND ENGINE

Description

The compound engine generally considered for aircraft propulsion consists of a conventional reciprocating engine, a steady-flow exhaust-gas turbine, and an auxiliary supercharger. A power plant of this type is diagrammatically illustrated in figure I-1. The engine exhaust gas is ducted to the turbine, which is provided with a nozzle for jet propulsion. The turbine drives the auxiliary supercharger and the excess turbine power is delivered to the engine shaft through gearing. An intercooler is provided for cooling the engine charge air after the auxiliary compressor. The shaft power of the system is converted to propulsive power by means of the propeller.

Engine Performance

The performance characteristics presented are for a compound engine comprising a four-row air-cooled engine of 4360-cubic-inch displacement and are based on the results of dynamometer-stand investigations of a multicylinder air-cooled engine of 2800-cubic-inch displacement. Turbine and auxiliary-supercharger efficiencies of 80 percent and an intercooler effectiveness of 50 percent were assumed. The efficiency of the gears between the turbine and the engine was taken as 95 percent.

One of the principal variables affecting the performance of the compound engine is the ratio of engine-exhaust (turbine-inlet) pressure to inlet-manifold pressure p_e/p_m. An increase in this ratio increases turbine power but decreases engine power. An optimum exhaust pressure exists for which the net performance of the system is a maximum. This effect is illustrated in figure I-2 where brake horsepower and brake specific fuel consumption (fuel only) are plotted against p_e/p_m for three altitudes and two power levels (approximately cruise and rated powers for the engine). The curves are for a flight speed of 400 miles per hour; however, their shape will not change greatly for other flight speeds.

The curves show that the minimum specific fuel consumption is obtained at a higher value of p_e/p_m than that corresponding to maximum power. A value of p_e/p_m of 1.0 represents a good compromise for all operating conditions and subsequent figures are based on this value.

The brake power increases initially with increasing altitude and reaches a maximum at an altitude between 30,000 and 50,000 feet. The eventual decrease in power at high altitudes is a result principally of the increasing charge temperatures in the intake manifold and consequent decreasing indicated power.

The specific fuel consumption decreases with increasing altitude principally as a result of increased turbine power. For altitudes between 30,000 and 50,000 feet, the improvement in efficiency is small and as altitude is further increased the specific fuel consumption will eventually pass through a minimum value. This effect is due mainly to the increased supercharger and engine friction power per pound of charge air as influenced by the constant ambient temperature above the tropopause.

Brake specific fuel consumptions of 0.43 and 0.35 pound per horsepower-hour for sea level and 30,000 feet, respectively, are indicated at $p_e/p_m = 1.0$ for the cruise condition (fig. I-2(a)). The fuel consumptions for the rated-power condition are necessarily higher because of the richer fuel-air mixture required.

The specific fuel consumption on a net-thrust-horsepower basis is plotted against flight speed in figure I-3 for the same altitudes and power levels as in figure I-2. The specific fuel consumption in this case includes both fuel and oil and the net thrust power on which it is based includes the propeller losses, cooling drag power, and exhaust-jet thrust power. The specific oil consumption, based on the brake power of the reciprocating engine only, was taken as 0.010 and 0.015 pound per horsepower-hour for the cruise- and rated-power conditions, respectively. The propeller efficiency for this and subsequent figures was assumed equal to 85 percent for Mach numbers up to about 0.6 and decreased at higher Mach numbers in accordance with test data. (See the appendix.) The lowest flight speeds plotted are those at which available ram pressure (0.9 of dynamic pressure) is just sufficient to maintain an average engine cylinder-head temperature of 450° F.

The fuel consumption for the cruise condition decreases, as in figure I-2, with increasing altitude for the range covered (fig. I-3(a)), and will, as previously stated, eventually reach a minimum value as the altitude is further increased. At rated power, the cooling drag power is very large at 50,000 feet and the altitude for minimum thrust horsepower specific fuel consumption is less than 50,000 feet (fig. I-3(b)). Values of specific fuel consumption of about 0.51 and 0.40 pound per net thrust horsepower-hour are indicated at sea level and 30,000 feet, respectively, for the cruise condition. The corresponding values for rated power are about 10 and 15 percent higher, respectively.

In order to facilitate comparison with the jet-propulsion power plants, which will be discussed in the subsequent parts of this report, the specific-fuel-consumption data of figure I-3 are shown in figure I-4 on the basis of net thrust. Net thrust specific fuel consumption is plotted against flight speed for the same altitudes and power levels as in figure I-3. The almost linear increase of thrust fuel consumption with speed is a direct reflection of the approximately constant thrust horsepower fuel consumptions shown in figure I-3. The variation with altitude is the same as before. At cruise power, a value of about 0.14 pound per hour per pound of thrust is obtained at 100 miles per hour for sea-level operation increasing to 0.57 pound per hour per pound of thrust at 500 miles per hour and 30,000 feet (fig. I-4(a)).

The net thrust in pounds per square foot of nacelle frontal area is plotted against flight speed in figure I-5 for the same conditions as figures I-3 and I-4. The frontal area used in calculating these curves is that of the four-row air-cooled engine assumed for the reciprocating-engine component of the compound engine plus allowance for nacelle clearance (engine diameter plus 3 in.). The thrust per unit frontal area could theoretically be increased by adding more rows of cylinders to an engine of the same diameter; however, four rows represent the maximum number currently used in large engines. The curves in figure I-5 are approximately right hyperbolas; therefore, doubling the flight speed halves the thrust. This variation is expected inasmuch as thrust horsepower is substantially constant over the speed range. The thrust varies with altitude in about the same manner as the brake power, which was previously discussed. For cruise power at 100 miles per hour and sea level, a thrust of about 360 pounds per square foot of frontal area is obtained decreasing to 87 pounds per square foot at 500 miles per hour and 30,000 feet (fig. I-5(a)). The corresponding values for rated power are about 68 and 58 percent higher, respectively (fig. I-5(b)).

The difference between net thrust and nacelle drag in pounds per square foot of nacelle frontal area is presented in figure I-6. The drag coefficient used for calculating nacelle drag was based on the result of wind-tunnel investigations and had a value of 0.056 up to a Mach number of 0.5, increasing to 0.065 at a Mach number of 0.7. (See the appendix.) Comparison of figures I-5 and I-6 shows that nacelle drag is practically negligible except at the higher portion of the speed range covered.

The thrust delivered by the compound engine per pound of engine weight is shown in figure I-7. The weight values used in this figure include: the constant weight of the reciprocating

engine and the auxiliaries; the weight of the auxiliary compressor, turbine, and intercooler, which varied with altitude; and the weight of the propeller, which varied with power, flight speed, and alti- tude. The curves in figure I-7 are similar to those in figure I-5 except for changes introduced by the variations in power-plant weight. For cruise power, a thrust of about 1.2 pounds per pound of engine weight is indicated at 100 miles per hour and sea level decreasing to about 0.27 pound per pound at 500 miles per hour and 30,000 feet (fig. I-7(a)). The corresponding values at rated power are about 50 percent higher (fig. I-7(b)).

Load-Range Characteristics

Accurate interpretation of power-plant performance in terms of airplane load-range characteristics is complicated and involves detailed considerations of airplane design, flight plan, and other factors. An approximate evaluation that can be used to illustrate the comparative performance of the different engines in the sub- sonic range of flight speed, however, can be made rather simply. The gross weight of the airplane per unit frontal area of the engine nacelle is given by

$$\frac{W_g}{A} = \frac{F - D_n}{A} \frac{L}{D} \tag{1}$$

where

W_g gross weight of airplane, pounds

A nacelle frontal area, square feet

F net thrust of engine, pounds

D_n nacelle drag, pounds

L/D lift-drag ratio of airplane without nacelles

The difference between net thrust and nacelle drag $F - D_n$ represents the thrust available for overcoming the drag of the rest of the airplane. Two cases are considered:

(1) Constant L/D: The value of L/D is taken as 18 at all flight conditions.

(2) Limiting wing load: The value of L/D is taken as 18 only at flight conditions where the resulting wing loading is

80 pounds per square foot or less; at other flight conditions the value of L/D is reduced to give a wing loading of 80 pounds per square foot.

Using the previously shown thrust minus nacelle-drag character-istics of the power plant (fig. I-6), the gross weight per unit frontal area was calculated for ranges of flight speed and alti-tude at a given power level of the engine.

The disposable load of the airplane per unit nacelle frontal area is taken as

$$\frac{W_d}{A} = \frac{W_g - W_s - W_e}{A} \tag{2}$$

where

W_d total disposable load, pounds

W_s structure weight, pounds

W_e power-plant weight (including propeller), pounds

The structure weight W_s including control equipment was assumed to be 40 percent of the gross weight, which is an average value for large conventional aircraft. From equation (2), it is seen that the disposable load can be obtained from the gross weight (equation (1)) and the power-plant weight.

The disposable load per pound of gross weight W_d/W_g is obtained by dividing equation (2) by equation (1).

The initial fuel rate in pounds per mile per square foot of nacelle frontal area is given by

$$\frac{w_f'}{A} = \frac{w_f}{F} \frac{F}{A} \frac{1}{V_o} \tag{3}$$

where

w_f' initial fuel rate, pounds per mile

w_f fuel flow, pounds per hour

V_o flight speed, miles per hour

Values of w_f/F and F/A can be obtained from figures I-4 and I-5, respectively, for various flight speeds and altitudes thus permitting calculation of w_f'/A. The initial fuel rate in pounds per mile per pound of gross weight w_f'/W_g can be obtained by dividing equation (3) by equation (1).

If the entire disposable load is considered to be fuel plus tank weight, a range factor KR (K × range) is obtained by the relation

$$KR = \frac{W_d}{W_g} \frac{W_g}{w_f'} \frac{1}{1.1} \text{ miles} \tag{4}$$

The factor 1/1.1 accounts for fuel-tank weight, which was assumed to be 10 percent of the fuel weight. For the compound engine, as previously mentioned, the fuel-weight also includes the lubricating-oil weight.

The correction factor K allows for deviations in flight plan and for the progressive reduction in gross weight and, hence, reduction in required fuel rate during the flight. The value of K is the ratio of the average to the initial fuel rate per mile per ton of initial gross weight. It may be computed for any desired flight plan. (See the appendix.) Illustrative values of K are given based on the Breguet range equation, which is derived on the assumption that L/D and specific fuel consumption (on a horsepower basis) remain constant during flight. Constant L/D requires a change in speed or altitude during the course of the flight, hence the operating speeds and altitudes to be presented correspond to initial values of these variables.

The load-range characteristics of the compound engine at cruise power for the case of constant L/D are shown in figure I-8(a) where the disposable load per pound of gross weight W_d/W_g is plotted against the initial fuel rate per ton of gross weight $2000 \, w_f'/W_g$ for a range of flight speeds at altitudes of 0, 15,000, 30,000, and 50,000 feet. A similar plot for the rated-power condition is given in figure I-8(b). Flight speeds below 200 miles per hour were not considered in figure I-8; speeds above 500 miles per hour were omitted because of the rapid increase in nacelle-drag power and decrease in propeller efficiency and engine thrust attending operation at the higher speeds.

At constant altitude, an increase in speed results in an increase in fuel rate and a decrease in disposable load. At

constant speed, an increase in altitude results in a decrease in
fuel rate and an increase in load up to an altitude of about
30,000 feet with subsequent decrease in load as altitude is further
increased. This effect is more marked at higher flight speeds.
For most of the flight conditions, cruise-power operation results
in slightly lower disposable loads and fuel rates than rated-power
operation. At low altitude-high speed conditions, however, cruise
power results in markedly lower disposable load and higher fuel
rate than rated power. The maximum values of disposable load for
the operating conditions covered are about 0.51 and 0.54 pound per
pound of gross weight for cruise- and rated-power operation,
respectively, and are obtained at 200 miles per hour over a range
of altitudes from sea level to 30,000 feet. Minimum initial fuel
rates of about 0.12 (cruise power) and 0.14 (rated power) pound
per ton-mile are indicated over a range of speeds at the higher
altitudes.

For the case of no pay load, that is, the entire disposable
load is fuel plus tank, the range factor KR at any speed and
altitude is obtained from the slope of a line drawn through the
origin and the point in question. The slope of such a line is
equal to the ratio of the disposable load to the initial fuel rate
(equation (4)). A scale is included in figure I-8 for convenience
in estimating KR; a curve of the variation of the correction
factor K with disposable load is given to permit calculation of
the actual range.

Maximum range is obtained at the operating point giving the
line of maximum slope, which is seen to be at 200 miles per hour
and 30,000 feet for both cruise- and rated-power operation
(fig. I-8). The value of KR for the cruise-power condition is
about 7400 miles (fig. I-8(b)); the value of K for the corre-
sponding disposable load is 0.74 from which the actual maximum
range is $\frac{7400}{0.74}$ or 10,000 miles. The maximum range is slightly less
for the rated-power condition; however, at the higher flight
speeds greater range is obtained for the rated-power than for the
cruise-power condition.

The allowable pay load for a specific range may also be
estimated from figure I-8. A line is drawn from the origin to the
desired range, for example KR equals 2000 miles (fig. I-8(a)).
Then the vertical distance from a given speed-altitude operating
point to the line is the pay load per pound of gross weight and
the rest of the vertical distance down to the abscissa is the fuel
load (plus tank) per pound of gross weight. The value of K is
obtained corresponding to this value of fuel load (plus tank) per

gross weight from the plot on the left-hand side of the figure.
(See the appendix.) The fuel load obtained in this manner is only
the amount required to cover the desired distance; reserve fuel
for emergencies would therefore be charged against the pay load.

Additional weight breakdown of the airplane can also be obtained
from figure I-8. Inasmuch as the figure is based on the assumption
of structural weight equal to 40 percent of the gross weight, the
vertical distance from an ordinate value of 1 down to 0.6 is the
structural weight per unit gross weight and the vertical distance
from 0.6 to any speed-altitude operating point represents the power-
plant (including propeller) weight per unit gross weight. The
improvement that is obtainable by a reduction in structural weight
or power-plant weight can be readily indicated on the figure. For
example, if the structural weight per unit gross weight were
reduced from 0.4 to 0.3 all the curves would be raised 0.1; for a
reduction in power-plant weight, each curve point would be raised
a percentage amount of the vertical distance between the point and
the structural weight line (the 0.6 ordinate in fig. I-8) equal to
the percentage reduction in power-plant (including propeller)
weight.

It is evident that where the operating point is close to the
structural weight line (0.6 in fig. I-8), for example, at a low
flight speed, there is little improvement to be gained by reduc-
tion in engine weight; however, where the operating point is
appreciably below the 0.6 ordinate, for example, at high flight
speeds, large improvement (large upward displacement of the oper-
ating point) can be achieved by the same percentage reduction in
engine weight.

The effect of a change in L/D can be indicated in figure I-8
for any given speed-altitude operating point by moving the point
along a line passing through the operating point and point X
(located at the coordinates abscissa = 0, ordinate = structural
weight line (0.6 in fig. I-8)) on the basis that the distance of
the operating point from point X is inversely proportional to the
value of L/D. The validity of this procedure can be ascertained
from examination of equations (1), (2), and (3). The effect of a
change in the ratio r of nacelle drag to engine thrust can be
indicated in a similar manner on the basis that the distance from
the operating point to the point X is inversely proportional to
1 - r. For example, at 500 miles per hour and 30,000 feet alti-
tude the values of cruise power thrust and nacelle drag are
approximately 90 and 20 pounds per square foot, respectively
(figs. I-5(a) and I-6(a)), hence 1 - r = 0.78. If the

nacelle drag were reduced to zero (completely submerged installation), $1 - r = 1$ and the effect of this change is obtained in figure I-8(a) by moving the operating point to point A where the distance XA is 78 percent of the distance from X to the original operating point.

The characteristics shown in figure I-8 apply only for the assumptions made in this analysis. The assumptions are representative of normal practice rather than of special applications. More than the 10,000-mile range indicated could be obtained, for example, by overloading the airplane, which would be equivalent to changing the assumption of structural weight equal to 40 percent of the gross weight. Lower flight speeds would also improve the range.

The L/D value of 18 (fig. I-8) would predicate extremely high wing loadings and attendant high take-off and landing speeds for airplanes designed to fly in the high speed-low altitude range. This condition is corrected in the limited wing-loading calculation wherein L/D was so adjusted as not to exceed a wing loading of 80 pounds per square foot over the range of operation covered. The following table lists the flight speeds and altitudes at which a wing loading of 80 pounds per square foot is compatible with an L/D value of 18:

Altitude, ft	0	15,000	30,000	50,000
Flight speed, mph	214	270	350	550

At higher speeds, L/D was reduced to values consistent with a wing load of 80 pounds per square foot; at lower speeds, L/D was maintained constant at 18 with attendant reduction in wing loading. The load-range characteristics for the assumption of limited wing loading are shown in figure I-9. Comparison of figures I-8 and I-9 shows that the high-altitude points and the low altitude-low speed points are not appreciably affected by the wing-loading limitation; therefore, the maximum range is still 10,000 miles. The sea-level high-speed characteristics are, however, seriously impaired, as is illustrated in figure I-10 where the sea-level curve from figure I-9(a) is superimposed on the curves of figure I-8(a). At 400 miles per hour, the initial fuel rate has been increased from 0.21 pound per ton-mile for a constant value of L/D of 18 to 0.39 pound per ton-mile for a constant wing loading of 80 pounds per square foot and the corresponding disposable load has been reduced from 0.36 to 0.15 pound per pound of gross weight.

Included in figure I-9 are several operating points for a turbosupercharged reciprocating engine. Point B (figs. I-9(a) and I-9(b)) is for a turbosupercharged engine operating at a

flight speed of 200 miles per hour and an altitude of 30,000 feet.
Points C and D (fig. I-9(a)) are for 500 miles per hour and alti-
tudes of 50,000 and 30,000 feet, respectively. The performance of
the turbosupercharged engine is obtained by assuming that all of the
engine exhaust gas passes through the turbine (that is, closed waste
gate) and that the engine exhaust pressure (turbine-inlet pressure)
is that which provides just enough turbine power to drive the auxil-
iary supercharger. The turbine and auxiliary supercharger effi-
ciencies are the same as used for the compound engine (that is,
80 percent).

At 200 miles per hour and 30,000 feet, the range of the turbo-
supercharged engine is about 75 percent of that for the compound
engine. A comparison of the curves for 500 miles per hour shows
that at a given altitude the compound engine gives considerably
greater range than the turbosupercharged engine.

                              Conclusion

The results of this study show that with the compound engine,
greatest range is obtained at low-to-moderate flight speeds and
moderate-to-high altitudes. The load-carrying capacity is good
at low speeds over a range of altitudes and economy is good over
a range of speeds at relatively high altitudes. Comparison of
the characteristics of the compound engine with those of the other
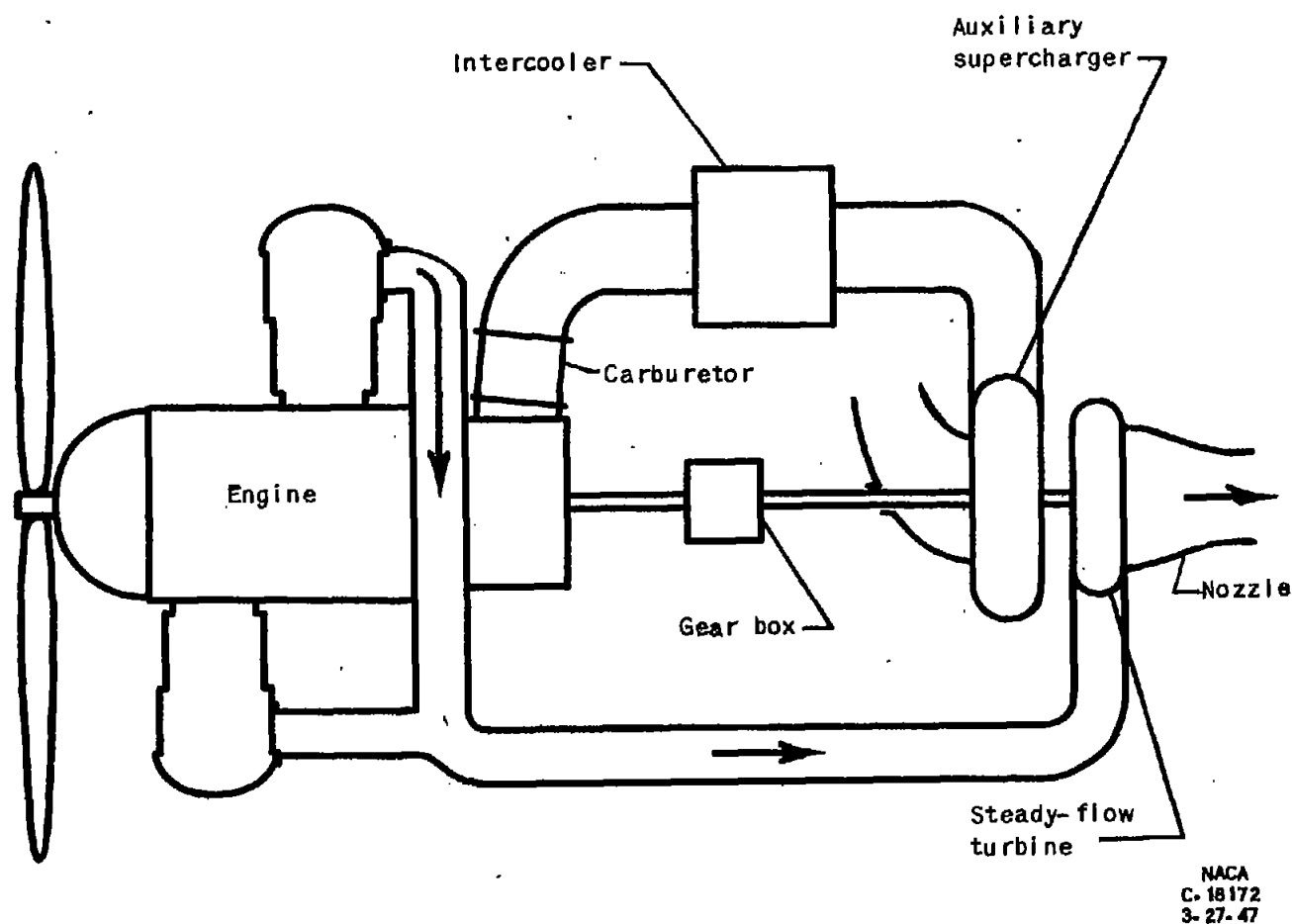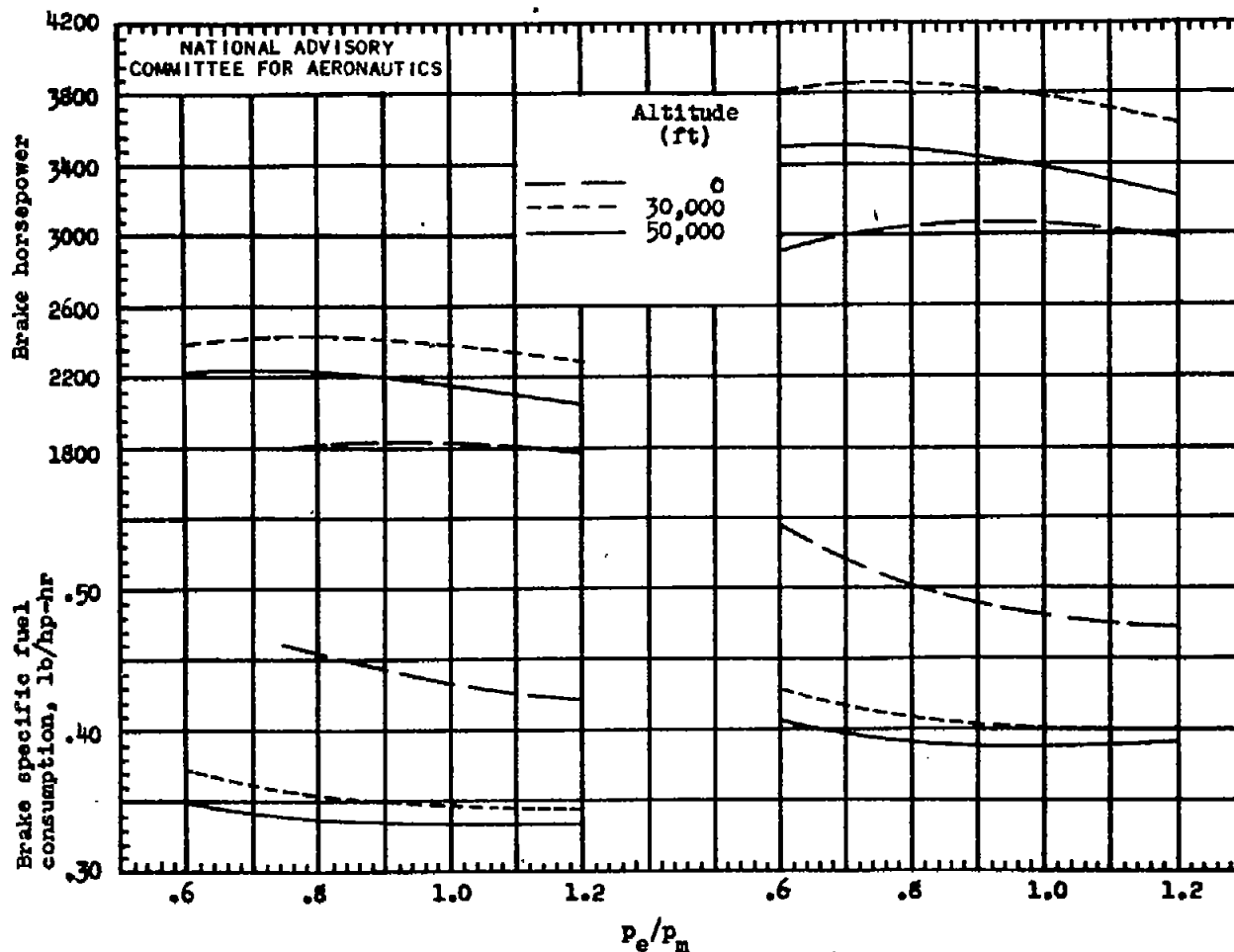power plants will be made in subsequent parts of this report.

Figure I-I. - Diagrammatic sketch of compound engine.

Fig. I-I

Fig. I-2

(a) Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.
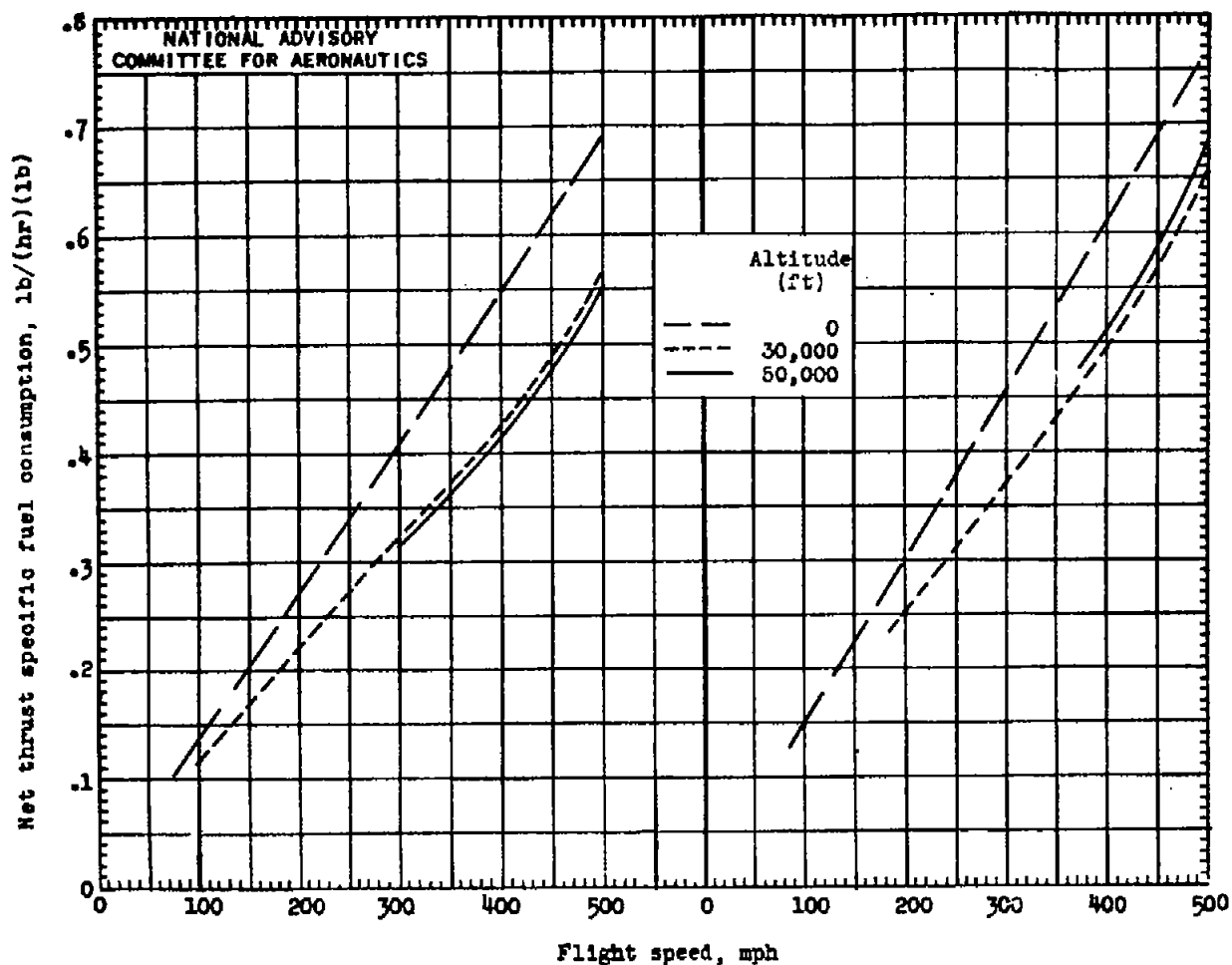
(b) Rated power: engine speed, 2600 rpm; inlet-manifold pressure, 50 inches mercury absolute; fuel-air ratio, 0.050.

Figure I-2. - Effect of exhaust pressure on brake horsepower and brake specific fuel consumption of compound engine. Flight speed, 400 miles per hour.
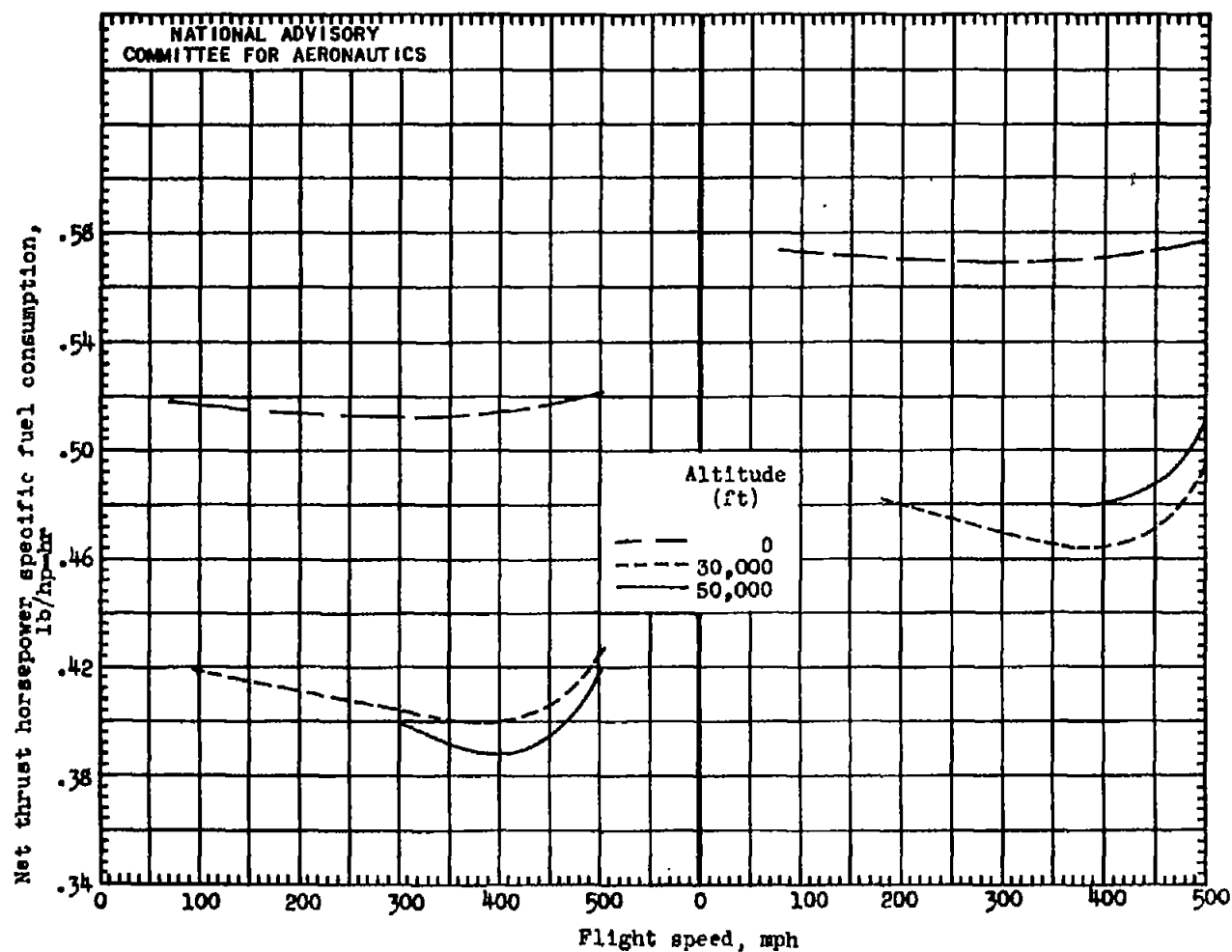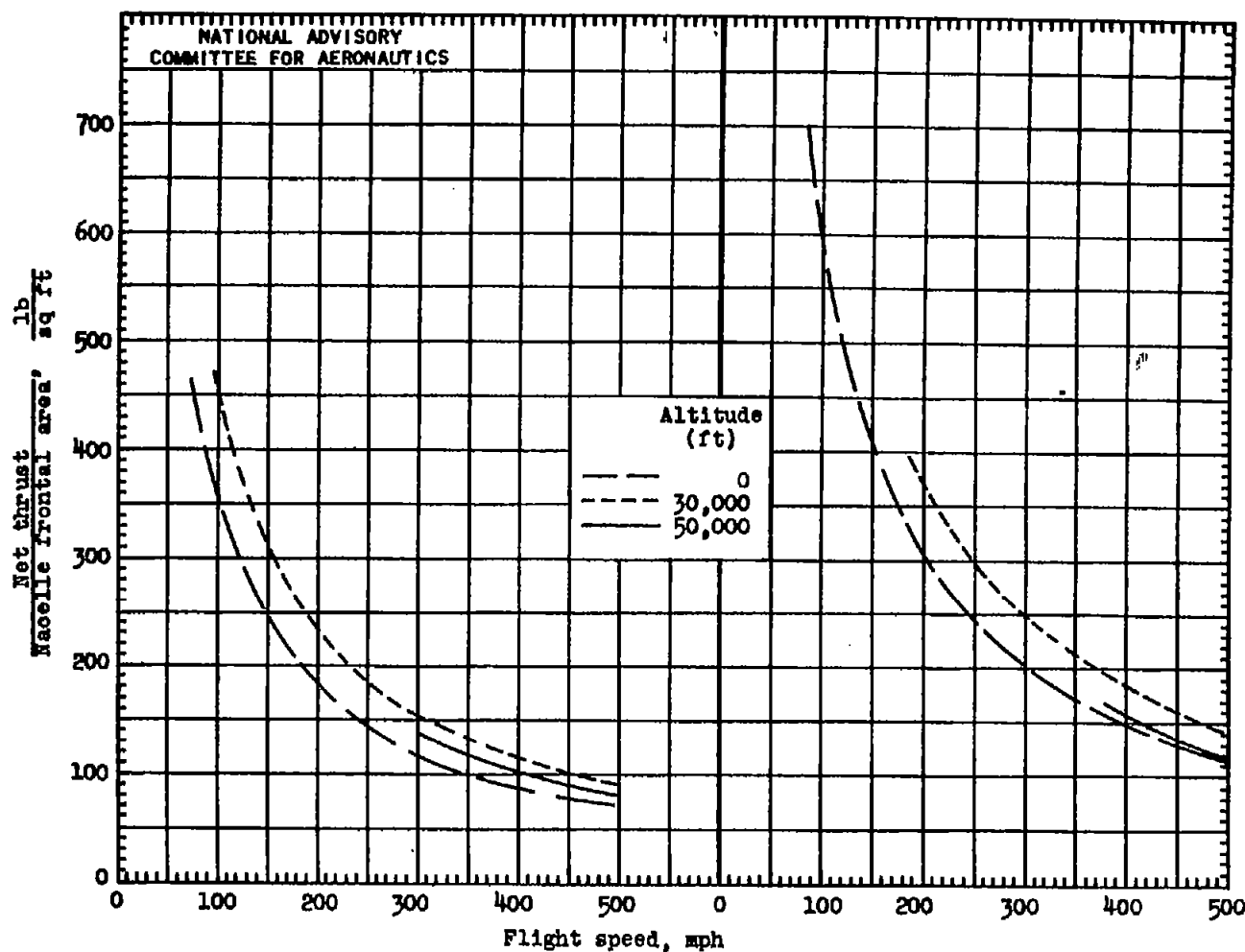
(a) Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.

(b) Rated power: engine speed, 2600 rpm; inlet-manifold pressure, 50 inches mercury absolute; fuel-air ratio, 0.080.

Figure I-3. - Variation of net thrust horsepower specific fuel consumption with flight speed and altitude for compound engine.

Fig. I-3

Fig. I-4

(a) Cruise power: engine speed, 2200 rpm;   (b) Rated power: engine speed, 2600 rpm;
     inlet—manifold pressure, 40 inches   inlet—manifold pressure, 50 inches
     mercury absolute; fuel—air ratio, 0.063.   mercury absolute; fuel—air ratio, 0.060.

Figure I-4. - Variation of net thrust specific fuel consumption with flight speed and altitude
for compound engine.

Fig. I-5

(a) Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.

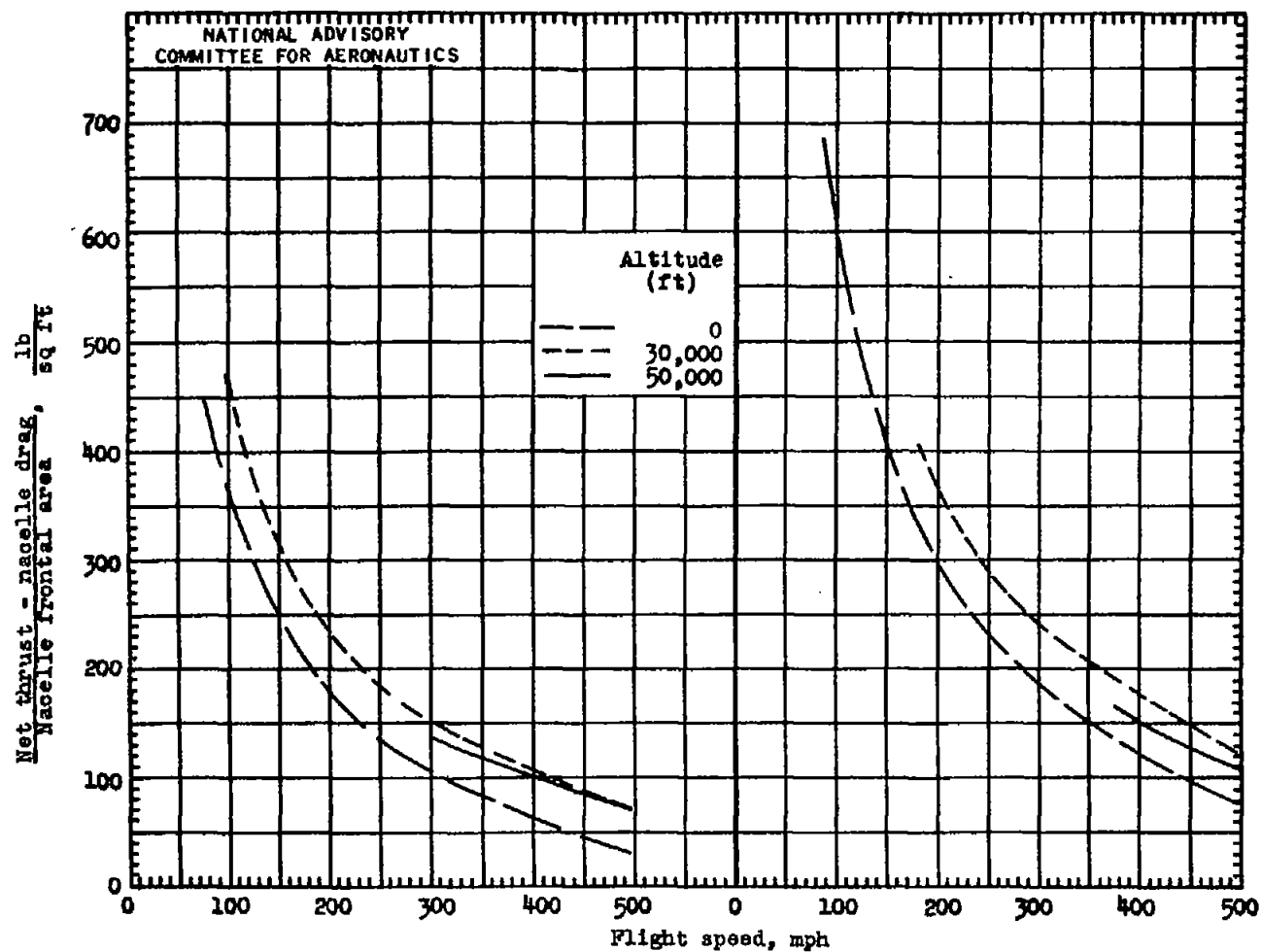(b) Rated power: engine speed, 2600 rpm; inlet-manifold pressure, 50 inches mercury absolute; fuel-air ratio, 0.080.

Figure I-5. - Variation of net thrust per unit nacelle frontal area with flight speed and altitude for compound engine.

Fig. I-6

(a) Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.

(b) Rated power: engine speed, 2600 rpm; inlet-manifold pressure, 50 inches mercury absolute; fuel-air ratio, 0.080.

Figure I-6. - Variation of net thrust minus nacelle drag per unit nacelle frontal area with flight speed and altitude for compound engine.

Fig. I-7

(a) Cruise power: engine speed, 2200 rpm;   (b) Rated power: engine speed, 2600 rpm;
     inlet-manifold pressure, 40 inches   inlet-manifold pressure, 50 inches
     mercury absolute; fuel-air ratio, 0.063.   mercury absolute; fuel-air ratio, 0.080.

Figure I-7. - Variation of net thrust per unit engine weight with flight speed and altitude for compound
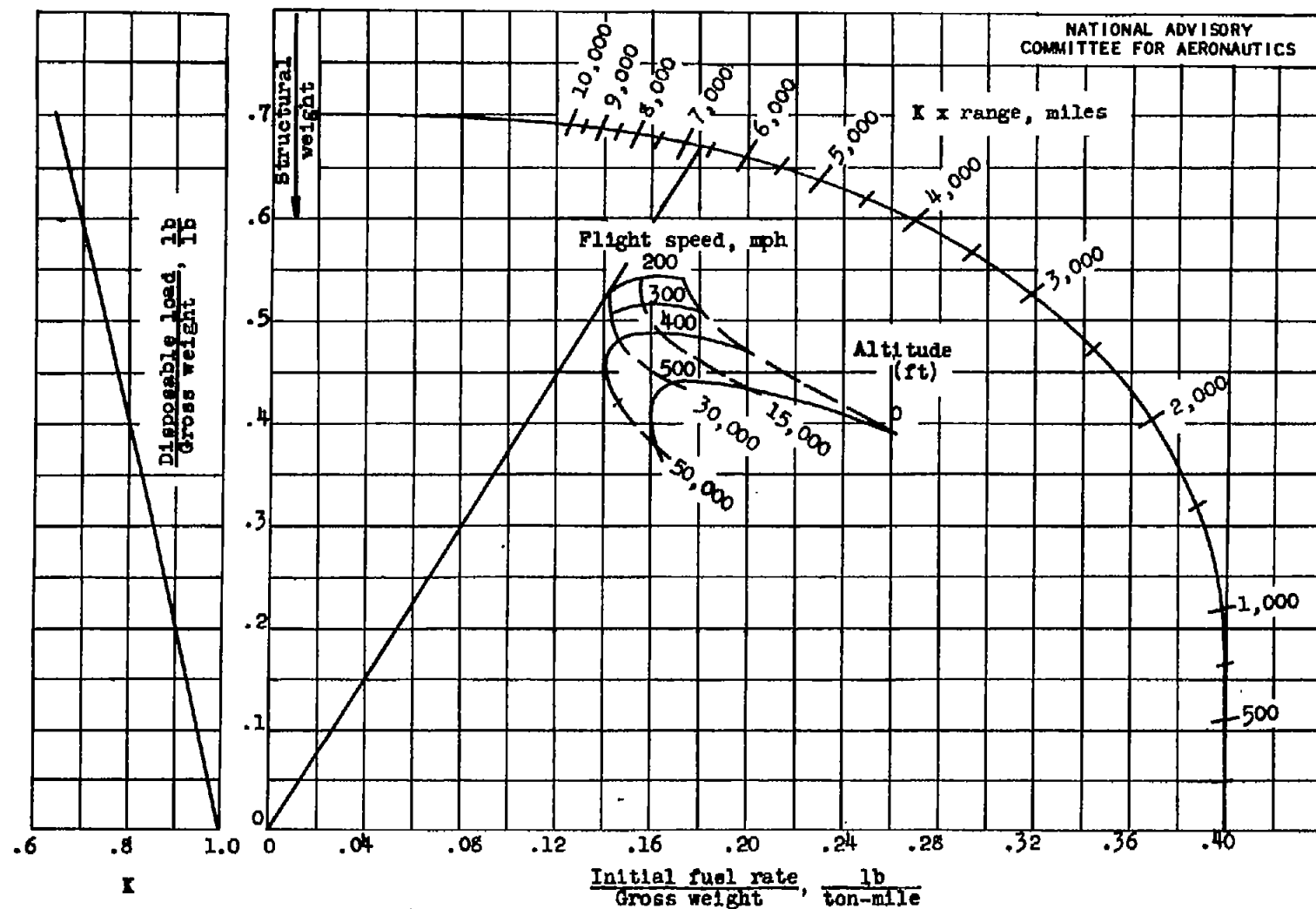engine.

Fig. I-8a
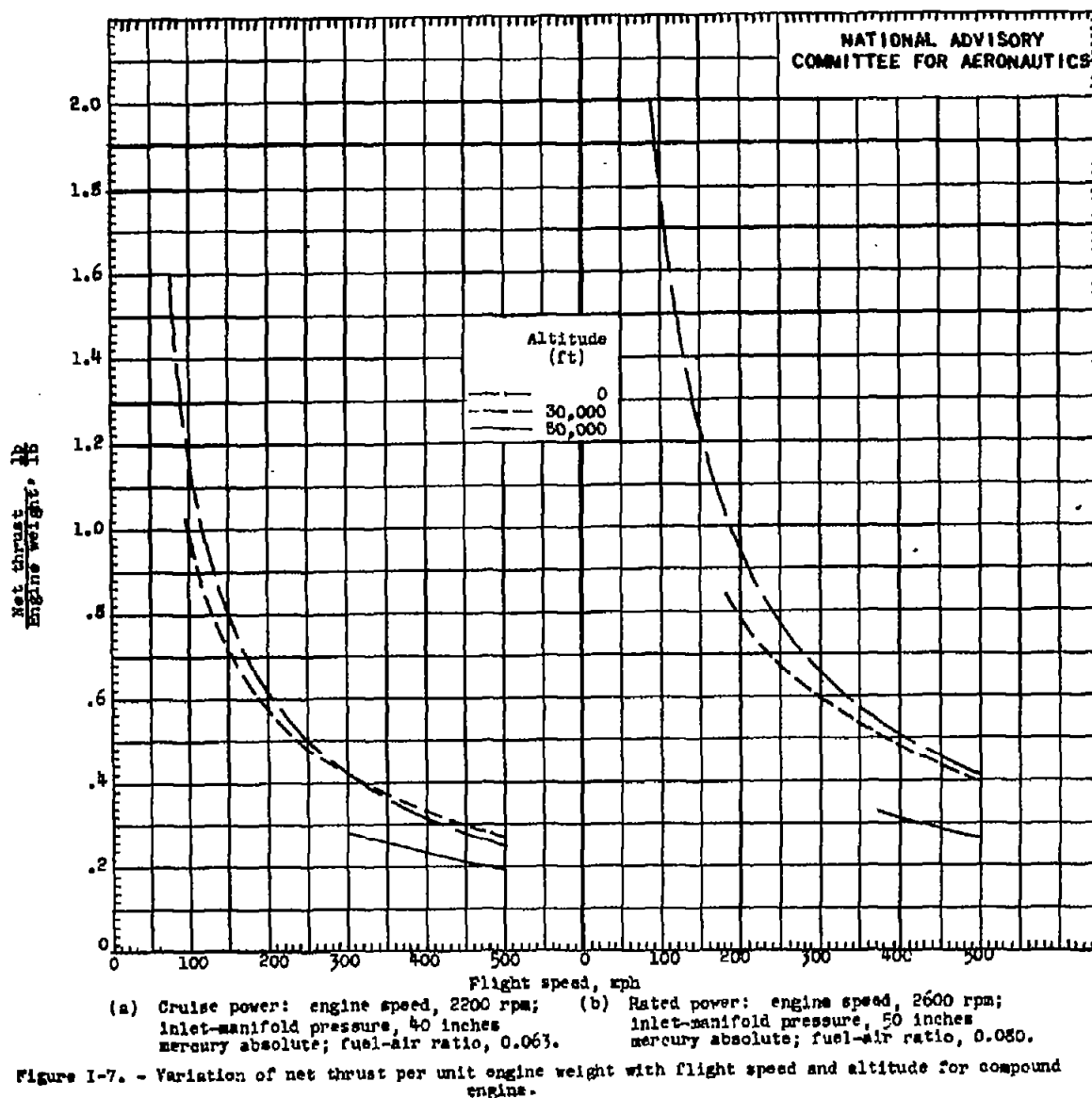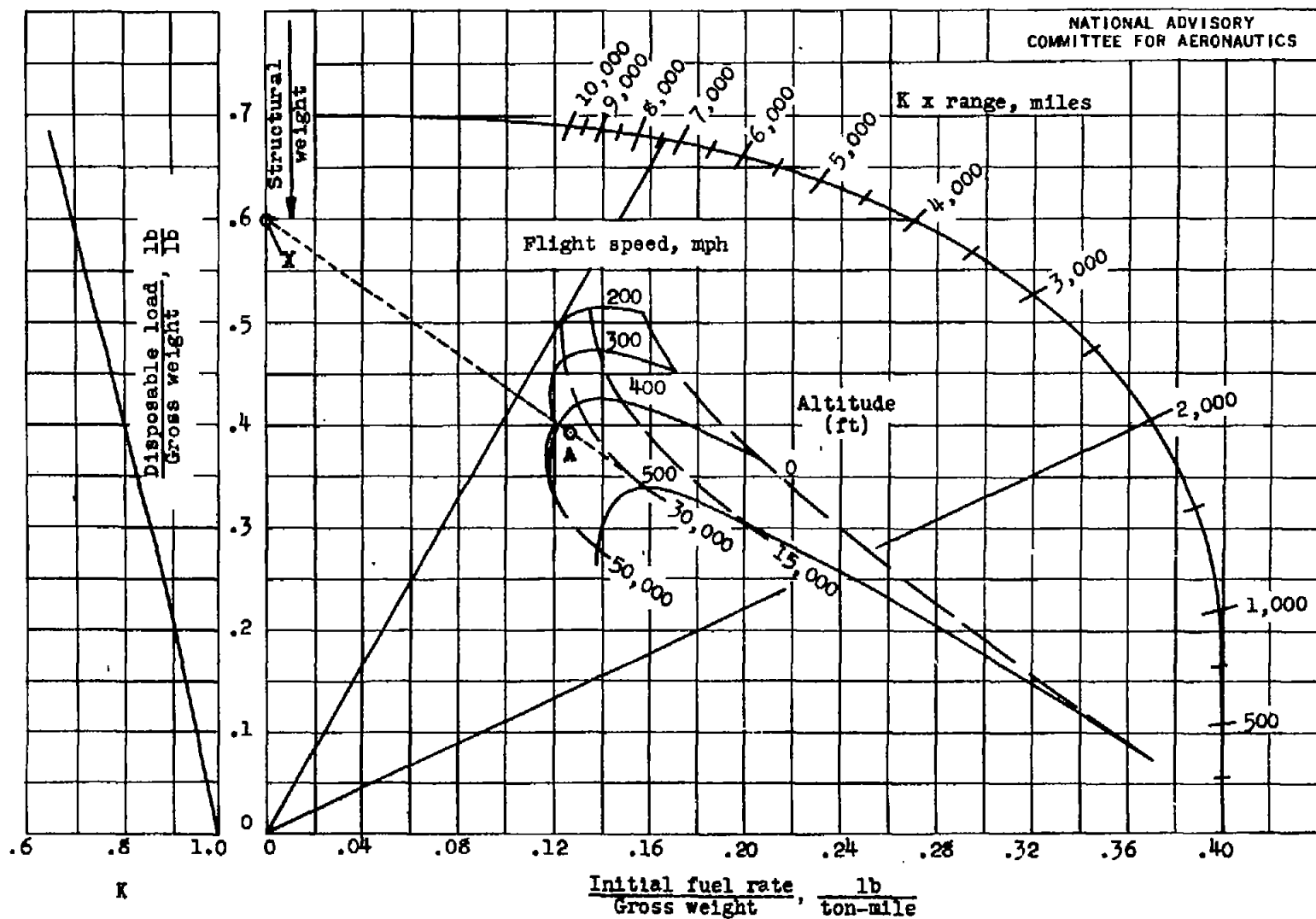
(a) Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.

Figure I-8. - Load-range characteristics of compound engine. Constant L/D, 18.

(b) Rated power: engine speed, 2600 rpm; inlet-manifold pressure, 50 inches mercury absolute; fuel-air ratio, 0.080.

Figure I-8. — Concluded. Load-range characteristics of compound engine. Constant L/D, 16.
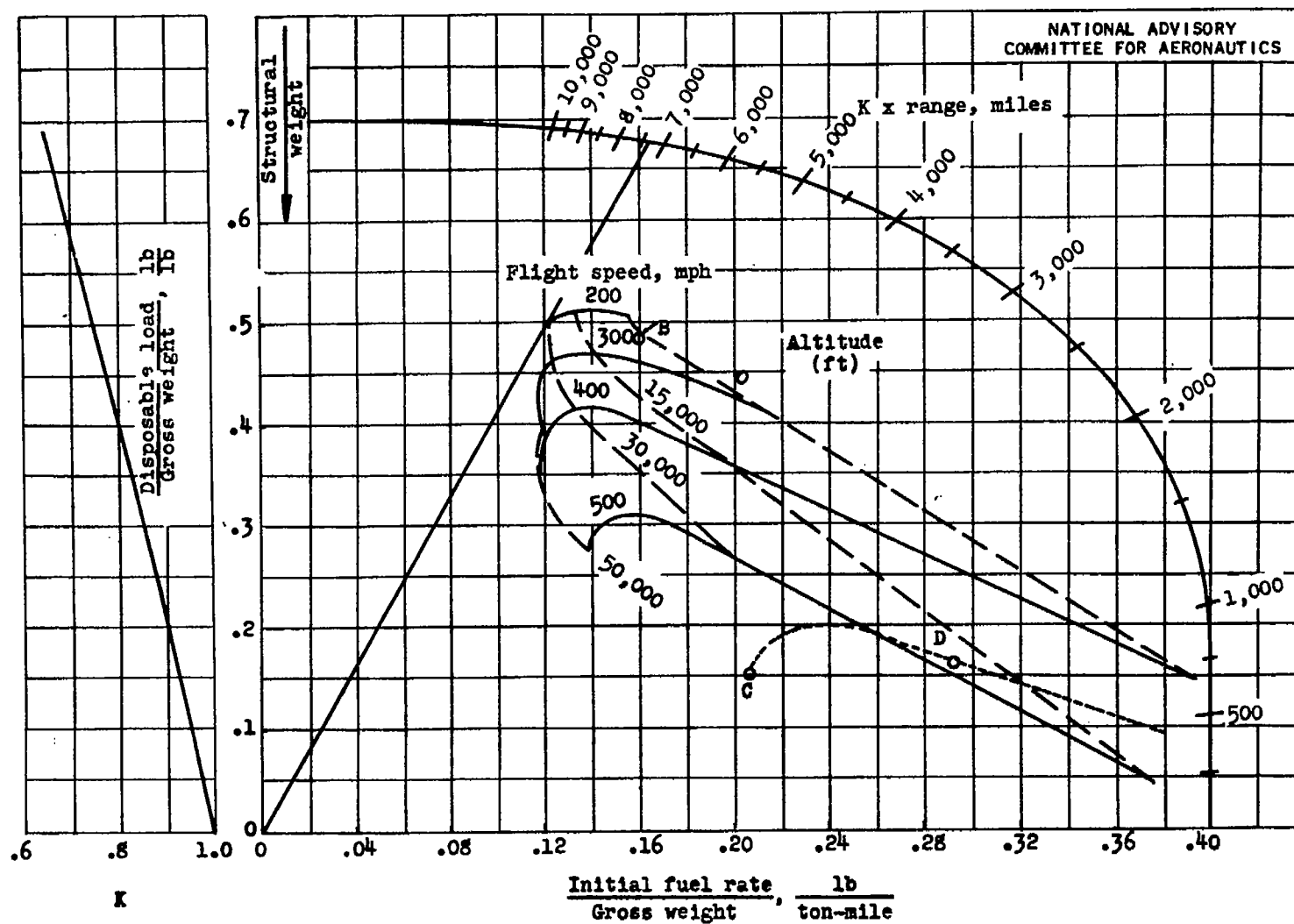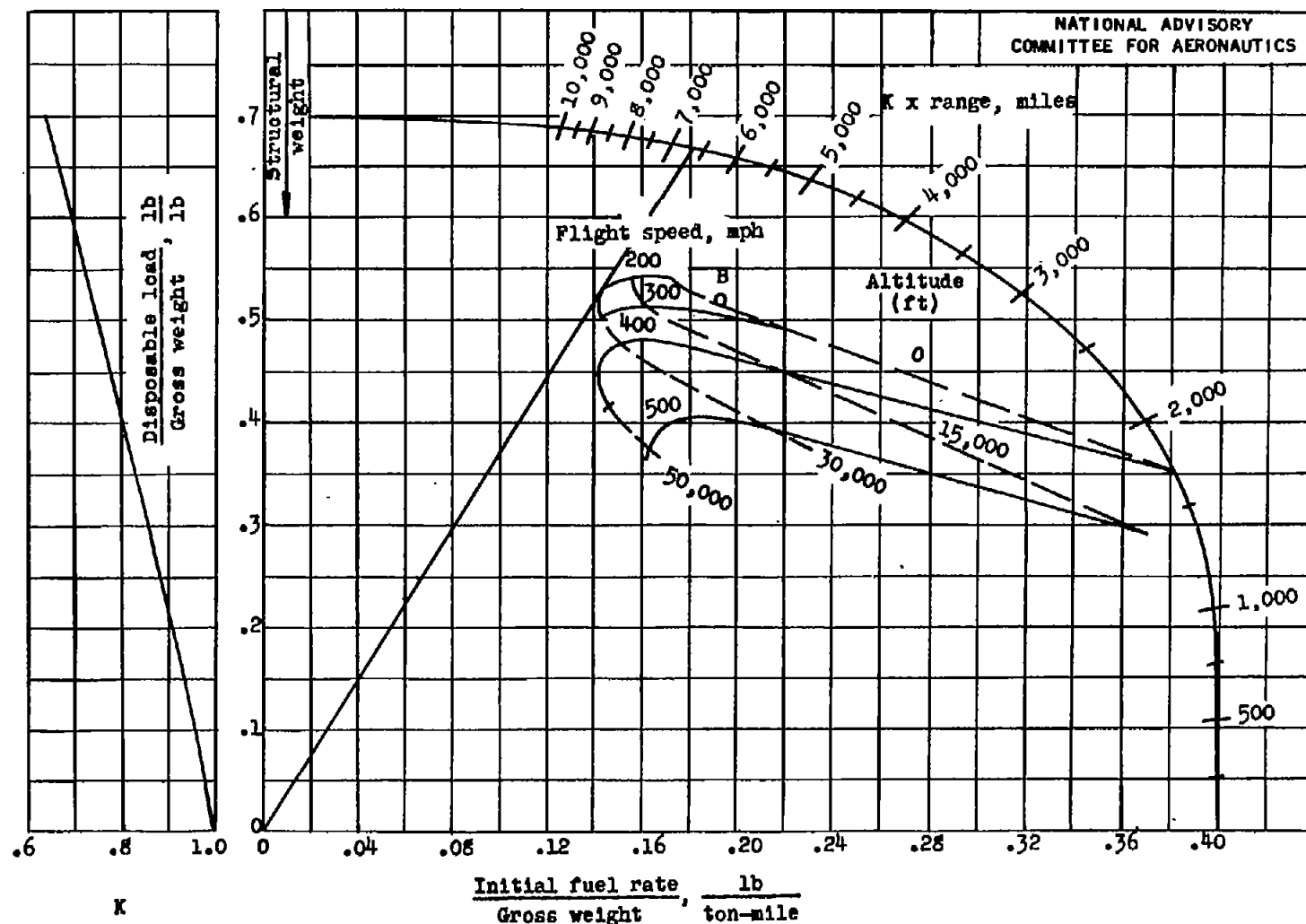
Fig. I-8b

Fig. I-9a

(a) Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.

Figure I-9. – Load-range characteristics of compound engine. Wing loading limited to 50 pounds per square foot.

(b) Rated power: engine speed, 2600 rpm; inlet-manifold pressure, 50 inches mercury absolute; fuel-air ratio, 0.080.

Figure I-9. – Concluded. Load-range characteristics of compound engine. Wing loading limited to 80 pounds per square foot.
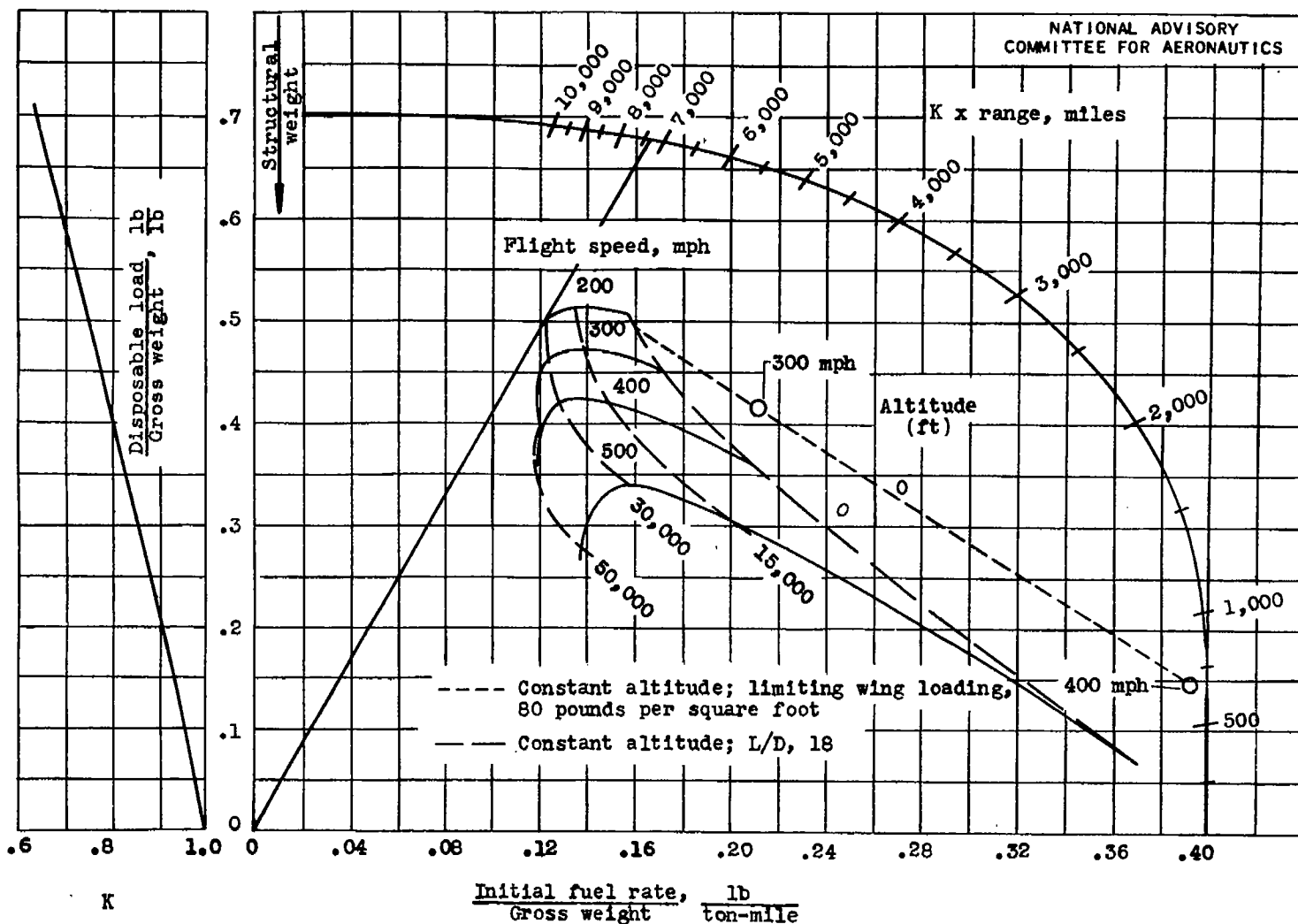
Fig. I-10

Figure I-10. - Load-range characteristics of compound engine. Constant L/D and limited wing loading. Cruise power: engine speed, 2200 rpm; inlet-manifold pressure, 40 inches mercury absolute; fuel-air ratio, 0.063.

II - THE TURBINE-PROPELLER ENGINE

Description

The gas turbine may be used to replace the reciprocating engine as a drive for a conventional propeller. A schematic diagram of such a plan is shown in figure II-1. A compressor inducts cold air from the atmosphere and compresses it to a high pressure. Fuel is mixed with the compressed air and burned and the gas is expanded through the turbine to approximately atmospheric pressure. The power created in expansion of the hot gas is more than required to compress the cold air and this excess power is utilized by a turbine-driven propeller and by a jet nozzle in back of the turbine.

Engine Performance

Presentation of the performance characteristics of this engine consists of: (1) an examination of the effects of some important design and operating parameters on the fuel consumption and power, (2) analysis of the performance of selected engines in an airplane in terms of load-carrying capacity and range, and (3) a comparison of the load-carrying capacities and ranges of aircraft powered by the gas turbine and the compound engine.

The effect on brake fuel consumption of increases in pressure ratio and cycle temperatures (ratio of turbine-inlet temperature to atmosphere temperature) is shown in figure II-2. In this figure the compressor and turbine efficiencies are assumed to be 80 percent and the combustion efficiency 95 percent.

Increases in turbine-inlet temperature decrease the fuel consumption provided the pressure ratio is properly increased. At the present limiting temperature of 1500° F at the turbine inlet, the temperature ratios at sea level and at 50,000 feet are indicated by points A and B, respectively, in figure II-2. At point A, corresponding to sea level, the optimum pressure ratio is shown to be between 8 and 16 or about 12. At point B, corresponding to an altitude of 50,000 feet, the optimum pressure ratio for minimum specific fuel consumption is above 16. At constant pressure ratio and the conditions presented in figure II-2, increases in turbine-inlet temperature resulted in increase in net work per pound of air.

The effect of changes in the efficiencies of the compressor and the turbine on fuel consumption is shown in figure II-3. For each temperature ratio and value of component efficiencies, the optimum

pressure ratio for minimum specific fuel consumption was chosen in
the manner shown in figure II-2. The efficiencies of the components
have a great effect on fuel consumption. For example, at sea level
with present limitations on cycle temperature (point A), an increase
in component efficiencies from 80 to 90 percent reduces the specific
fuel consumption from 0.65 to 0.38 pound per brake horsepower-hour.
Thus it appears that considerable variation in the fuel consumption
of turbine-propeller engines may be obtained by small changes in
compressor and turbine efficiencies and in turbine-inlet tempera-
tures, and any specific choice of these values for purposes of
comparing turbine-propeller engines with other engines is subject
to wide latitude. For the remainder of this study the following
efficiencies have been assumed: compressor, 85 percent; turbine,
90 percent; combustion chamber, 95 percent; intake diffuser,
90 percent; and jet nozzle, 94 percent. A pressure ratio of 12 is
assumed except where otherwise noted, and a turbine-inlet tempera-
ture of 1500° F is used. The division of power between the propeller
and the jet was chosen to give maximum thrust power for each operating
condition.

In the analysis of the effects of flight speed and altitude upon
specific fuel consumption, the efficiency of the propeller must be
considered. Fuel consumption is on the basis of pounds of fuel per
net thrust horsepower-hour. Figure II-4 shows that increasing speed
decreases the fuel consumption slightly until severe losses in
propeller efficiency at high speed cause an increase in fuel con-
sumption. Increased altitude reduces the fuel consumption because
a higher temperature ratio is permitted, as shown in figure II-3.
Under the conditions assumed, the specific fuel consumption lies
between 0.44 and 0.54 pound per net thrust horsepower-hour at speeds
below 500 miles per hour (fig. II-4).

The power characteristics, as well as fuel consumption, must
be evaluated before comparative studies of the engines can be made.
The power-weight ratio (including propeller), as expressed in terms
of thrust-weight ratio of a turbine-propeller engine, is shown in
figure II-5. For this figure the lowest weight-horsepower ratio at
90 percent of maximum power attained in test from available litera-
ture on turbine-propeller engines was used. This ratio at static
sea-level conditions was corrected to account for variations in
flight speed, altitude, and pressure ratio. The correction was made
by computing the change in work output per pound of air, change in
air capacity of the engine, and change in the weights of the engine
parts. The air capacity was corrected by assuming that the Mach
number of the air entering the compressor was constant. The weights
of the components were corrected for changes in compression ratio by

assuming that a portion of the engine weight is independent of the pressure ratio, and that the remaining portion is proportional to the number of compressor and turbine stages. According to these calculations, the ratio of the weights of engines with compression ratios of 12 and 5 was 1.4.

The weight-horsepower ratio at static sea-level conditions of the engine without propeller used for this analysis based on the lightest current engine per unit power, the weight-horsepower ratio of a representative or average current engine, and an estimated weight-horsepower ratio obtained by adding additional turbine and gearing weight to a turbojet engine are shown uncorrected and corrected to a pressure ratio of 12 in the following table. The equivalent horsepower was computed by adding to the shaft horsepower the quotient obtained by dividing the static thrust of the exhaust jet by 4.

Engine	Compression ratio	Engine weight (lb/bhp)			
		Continuous-rated static equivalent power at sea level		Maximum static equivalent power at sea level	
		Observed	Compression ratio, 12	Observed	Compression ratio, 12
Lightest surveyed (used in the analysis)	5	0.734	1.03	0.66	0.927
Representative	6	.906	1.17	.815	1.05
Converted turbojet	4	.56	.73	.50	.66

The engine chosen for the analysis had a weight-horsepower ratio of 1.03 pounds per brake horsepower with a compression ratio of 12 at static sea-level conditions. The selection of this weight is subject to wide latitude because of the uncertainty in the accuracy of the estimate of the effect of compression ratio on engine weight. Further, the analysis of the converted turbojet engine indicates the possibility of considerable reduction in weight-horsepower ratio.

Figure II-5 shows that high thrust-weight ratios are obtained
at low flight speeds at sea level, but the thrust decreases rapidly
with increases in speed and altitude. The rapid loss in thrust with
increase in altitude is the first significant difference between the
turbine-propeller engine and the compound engine. The compound
engine is assumed to be supercharged sufficiently to maintain mani-
fold pressures required at sea level up to altitudes as high as
considered in this study (50,000 ft). Consequently, the turbine-
propeller engine, which produces more thrust than the compound
engine for a given weight at sea level, will at certain altitudes
produce less thrust than the compound engine. Figure II-6 compares
the effects of altitudes upon the thrusts of these two engines with
the compound engine operating at cruise power.

The thrust per unit frontal area is of importance when the engine
is quite large in proportion to its power and when high flight speeds
are considered. Figure II-7 shows the thrust per unit frontal area
of the hypothetical engine at altitudes from sea level to
50,000 feet and flight speeds from 100 to 500 miles per hour.
Increases in both altitude and speed decrease the thrust per unit
engine frontal area. These curves are representative of some
existing turbine-propeller engines. Studies of turbojet-engine com-
ponents indicate that the thrust per unit engine frontal area could
be increased at a possible cost of increased weight and fuel
consumption.

Load-Range Characteristics

The load-carrying capacity and the range of an airplane are
affected by the fuel consumption and the engine weight. Charts
showing disposable load, fuel rate per ton-mile, and range for
various speeds and altitudes are shown in figure II-8. Figure II-8(a)
shows the load-range characteristics when the lift-drag ratio is
maintained at 18. In figure II-8(b), the wing loading is limited to
80 pounds per square foot. A maximum lift-drag ratio of 18 was
chosen for conditions where this lift-drag ratio could be attained
without exceeding a wing loading of 80 pounds per square foot.
Nacelle drag was deducted from engine thrust. Comparison of fig-
ures II-8(b) and II-8(a) shows that for high-speed service, con-
siderable loss in performance results from the use of wings large
enough to limit wing loading to 80 pounds per square foot, and that
better high-speed performance at low altitudes would be achieved by
using smaller wings and assisted take-off.

Figure II-8 shows that the greatest disposable load and lowest initial fuel rate are obtained at the lowest flight speed considered. The range, obtained by drawing a line from the origin through the selected operating point and extending this line to the scale of $K \times$ range (fig. II-8(b)), shows a maximum $K \times$ range of 6560, which is less than that for the compound engine. At flight speeds above 300 miles per hour, the disposable load is approximately constant with variation in altitude up to about 30,000 feet. Above this altitude the disposable load falls rapidly.

An engine with a pressure ratio of 12 was assumed for figure II-8. Selection of a pressure ratio giving maximum range or maximum disposable load at a specified range is complicated and has been worked out for only two extreme cases. An increase in pressure ratio up to 12 decreases the thrust per pound of engine weight but improves the fuel consumption. The effects of this phenomenon are illustrated in figure II-8(b). In one example, at a flight speed of 200 miles per hour at sea level decreasing the pressure ratio from 12 to 5 increased the disposable load slightly at a cost of considerable increase in fuel consumption and loss in maximum range. In another case, at an altitude of 50,000 feet and a speed of 500 miles per hour, the weight of the engine with a pressure ratio of 12 is so great that little capacity is left for disposable load. In this case reduction of the compression ratio from 12 to 5 reduces the engine weight sufficiently to increase the $K \times$ range from 130 to 1040 miles. Thus it is shown that the optimum pressure ratio for a given type of aircraft service cannot be computed from variations in engine characteristics alone, but the type of service must be considered. Even at a specified flight speed and altitude, the optimum pressure ratio varies with specified range.

Point A in figure II-8(b) represents an existing turbine-propeller engine with a compression ratio of 5 operating at 500 miles per hour at an altitude of 33,000 feet, and again shows that a compression ratio lower than 12 provides greater disposable load at high speed at the cost of a higher fuel rate.

Analysis of the weight of a turbojet engine converted for producing shaft power with an additional turbine and a gear box indicated that the weights of turbine-propeller engines might be reduced 30 percent without increasing cycle temperatures. An additional curve is therefore presented in figure II-8(b) to show the performance of the turbine-propeller engine at 500 miles per hour if future progress reduces engine and propeller weight 40 percent. An increase in disposable load of 41 percent and an increase in $K \times$ range of 41 percent would result at an altitude of 30,000 feet.

An example of the effects of nacelle drag on performance is also shown in figure II-8(b) for a flight speed of 500 miles per hour at an altitude slightly above 30,000 feet. The effect of a change in the ratio r of nacelle drag to engine thrust can be indicated on the basis that the distance from the operating point to the point X is inversely proportional to 1 - r. In the case considered, the value of r is 0.21 (taken from fig. II-5), and if drag were eliminated the operating point would move along the broken line to point B. This elimination of the nacelle drag increases the disposable load and K × range   23 and 58 percent, respectively.

Comparison of Turbine-Propeller Engine

and Compound Engine

Inasmuch as estimates of the performance of the turbine-propeller engine and the compound engine are available, a comparison of the load-carrying capacities and ranges of airplanes powered by these engines may be made. Data from figure II-8(b) for the turbine-propeller engine are compared with data from a similar figure presented for the compound engine. The results are shown in figure II-9. The broken line in the center of the field separates the regions where the turbine-propeller engine having a pressure ratio of 12 and the compound engine show the greater load-carrying capacities at the specified speeds and fuel rates per ton-mile, respectively. The turbine-propeller engine shows somewhat greater load-carrying capacities at low flight altitudes and speeds than the compound engine; the compound engine shows greater load-carrying capacities at the various speeds at high altitudes.

The ability of the compound engine to carry disposable loads greater than those of the turbine-propeller engine at high flight speeds and altitudes is a result of the supercharging accomplished in the compound engine. As shown in figure II-6, the turbine-propeller engine produces more thrust per unit weight than the compound engine at sea level; this difference disappears at about 30,000 feet, and at higher altitudes the compound engine is more powerful. Furthermore, the minimum specific fuel consumption of the compound engine is lower. Consequently, as shown in figure II-9, at high flight speeds of approximately 500 miles per hour, aircraft powered by the compound engine may fly at high altitudes to permit operation at the economical maximum lift-drag ratio with engines no heavier than the turbine-propeller engines required at a lower and less economical altitude and will therefore have the greatest range.

These facts indicate that the weight-horsepower ratio of the turbine-propeller engine (neglecting propeller weight) with a pressure ratio of 12 at static sea-level conditions must be less than the 1.03 pounds per equivalent brake horsepower chosen for this analysis if it is to compete with the compound engine at a flight speed of 500 miles per hour.

The effect of engine weight is again shown in figure II-10. In this case the equivalent static sea-level values of pounds of engine weight per horsepower are shown and the comparison is made for a flight speed of 500 miles per hour. This figure shows that a turbine-propeller engine with a pressure ratio of 12 must have a static sea-level weight-horsepower ratio of 0.4 pound per horsepower if it is to have as great a maximum range as the compound engine at 500 miles per hour. Heavier engines could be permitted at less than maximum ranges.

The pay loads of the two engines are compared in figure II-11. The weight-horsepower ratio of the turbine-propeller engine at static sea-level conditions was assumed to be 1.03 pounds per horsepower, a value obtained by adjusting to a pressure ratio of 12 the weight of the lightest engine that has been tested and for which data are available. At each range and flight speed the aircraft were assumed to fly at altitudes providing the greatest load-carrying capacity. Figure II-11 shows that the turbine-propeller engine can carry greater loads than the compound engine for ranges up to 2900 miles at 200 miles per hour, and that this range of equal load-carrying capacity decreases with increasing airplane speed until at about 500 miles per hour the compound engine shows greater load-carrying capacity at all ranges. If the flight altitude is limited to 20,000 feet, the turbine-propeller has the greater load-carrying capacity at attainable ranges and speeds.

Conclusion

In this analysis the fuel consumption given for the turbine-propeller engine is optimistic in regard to present practice. The weight of the engine used in this study was obtained by correction of the weight of an existing turbine-propeller engine to a higher compression ratio. Under these conditions the range estimates show that the gas-turbine engine with high pressure ratio may provide long ranges at low speeds and moderate altitudes. The compound engine, as a result of its lighter weight per unit thrust at high altitudes, provides greater range than would be obtained from the turbine-propeller engine at high speeds. Analysis indicates the possibility of utilizing lighter turbine-propeller engines per unit thrust than assumed and this reduction would be necessary if the turbine-propeller engine is to provide a range equal to that of the compound engine at a flight speed of 500 miles per hour.
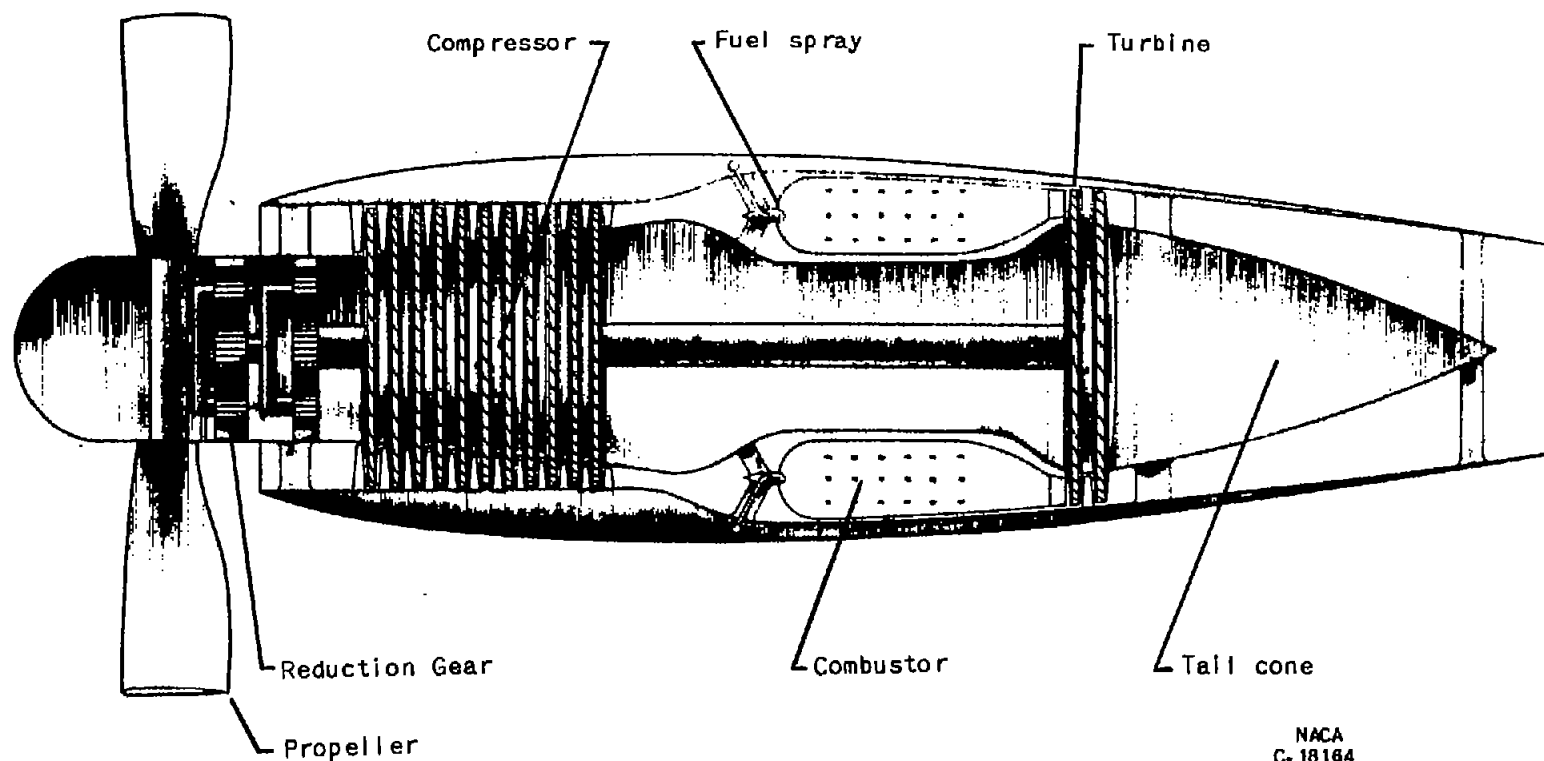
Figure II-I. - Diagrammatic sketch of turbine-propeller engine.

Fig. II-I

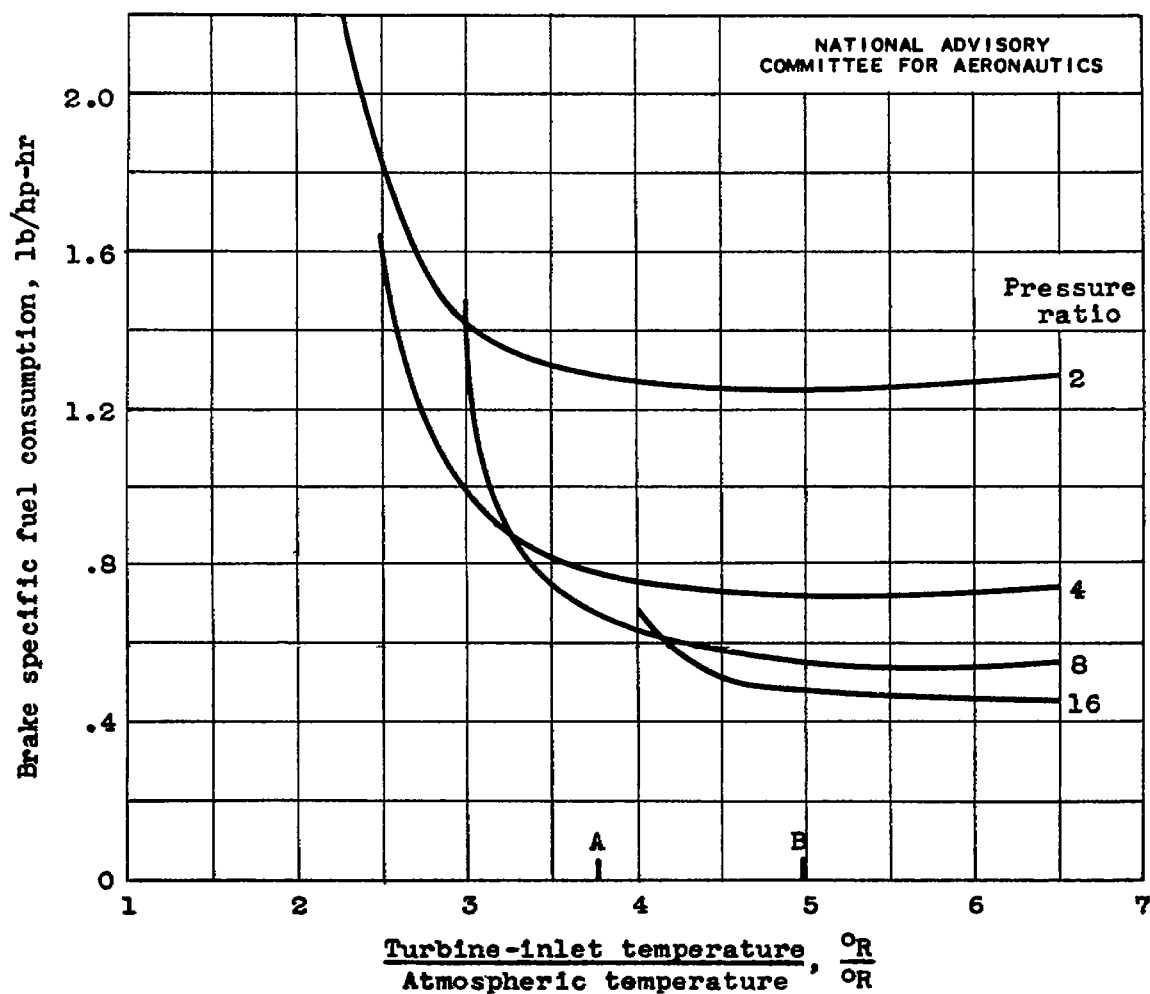Fig. II-2   NACA TN No. 1349

Figure II-2. - Effect of pressure ratio and cycle temperatures on brake specific fuel consumption of turbine-propeller engine. Compressor and turbine efficiencies. 80 percent; combustion efficiency, 95 percent.
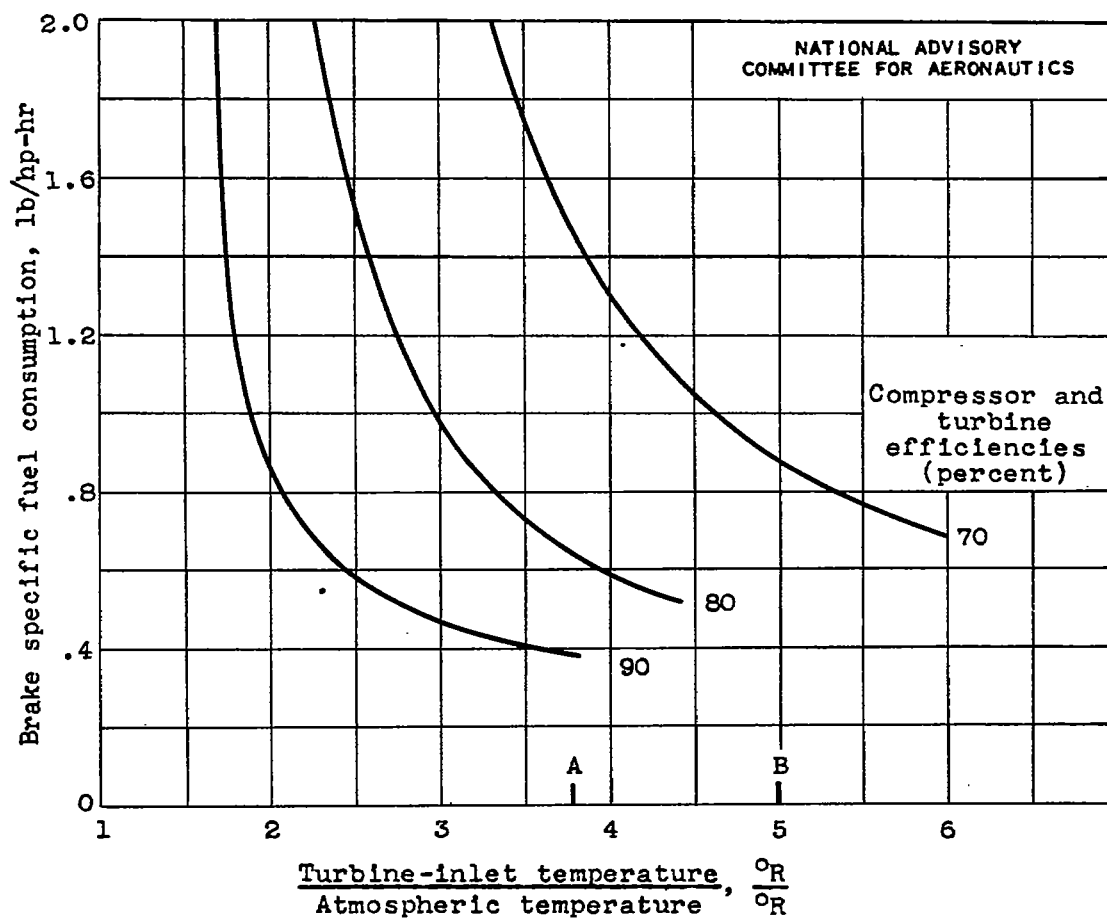
Figure II-3. - Effect of compressor and turbine efficiencies on brake specific fuel consumption of turbine-propeller engine. Pressure ratio for minimum fuel consumption.
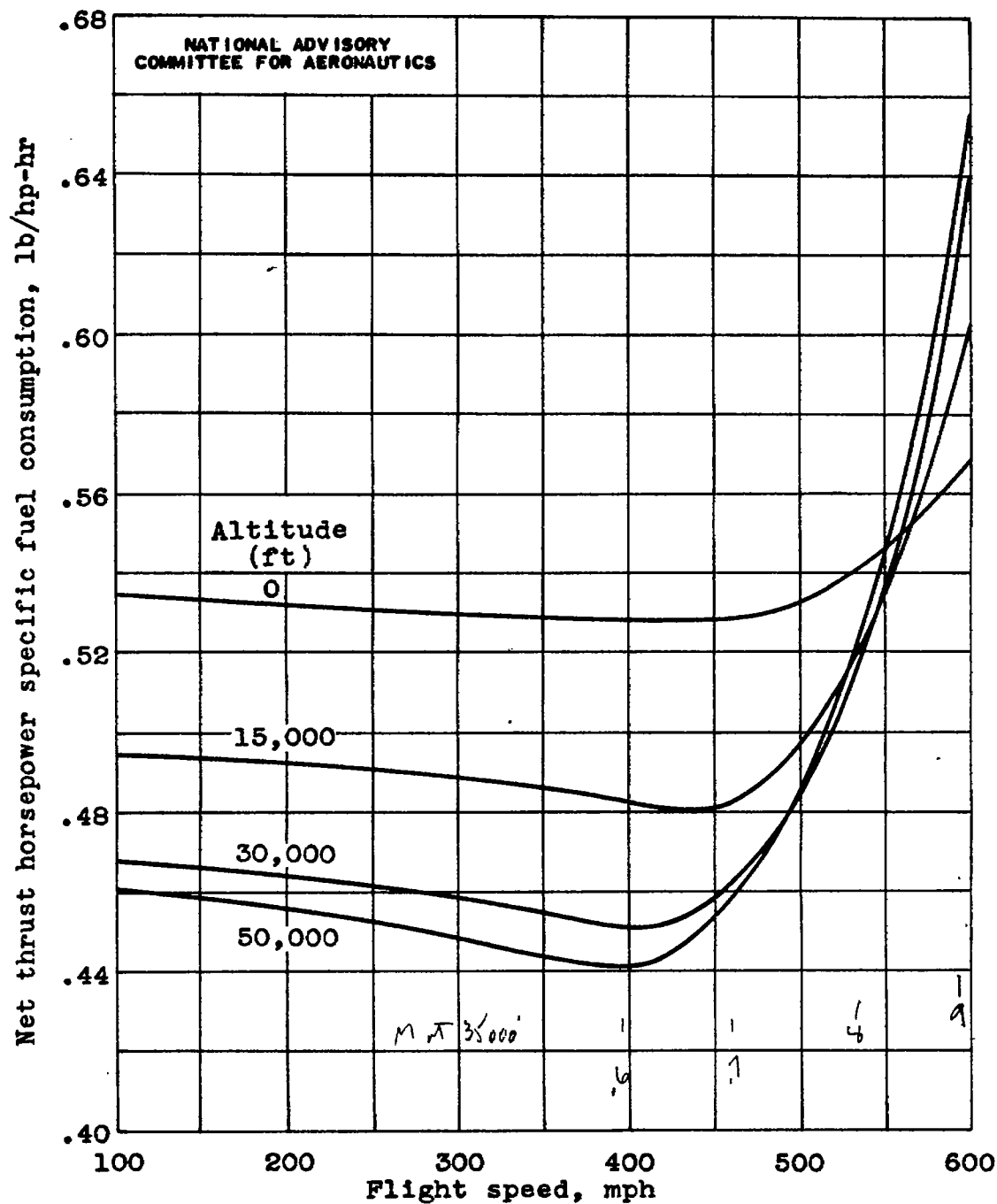
Fig. II-4

NACA TN No. 1349

Figure II-4. - Variation of net thrust horsepower specific fuel consumption with flight speed and altitude for turbine-propeller engine. Turbine-inlet temperature, 1500° F.
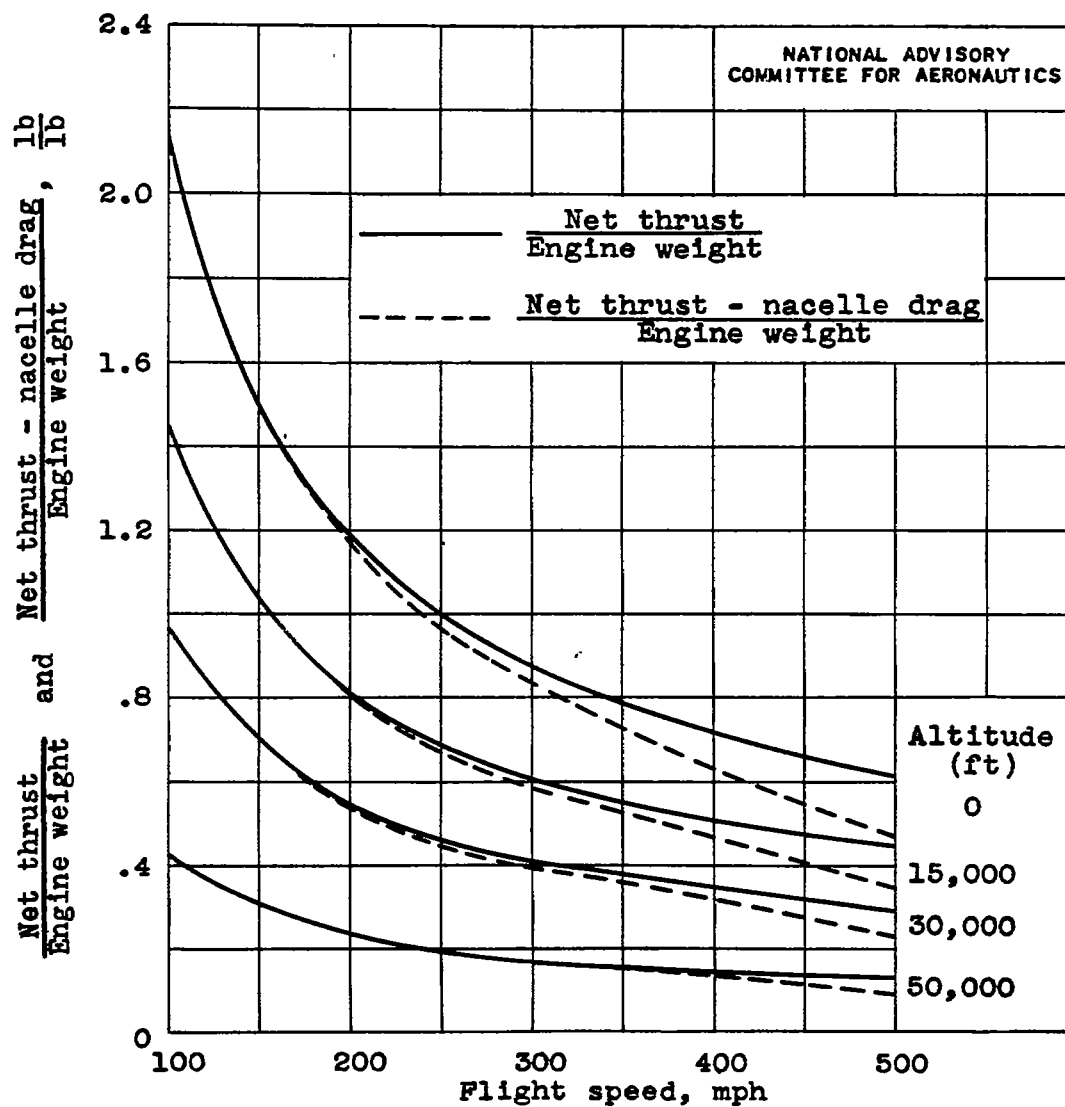
755-B

Figure II-5. - Variation of net thrust-weight ratio with flight
speed and altitude for turbine-propeller engine.

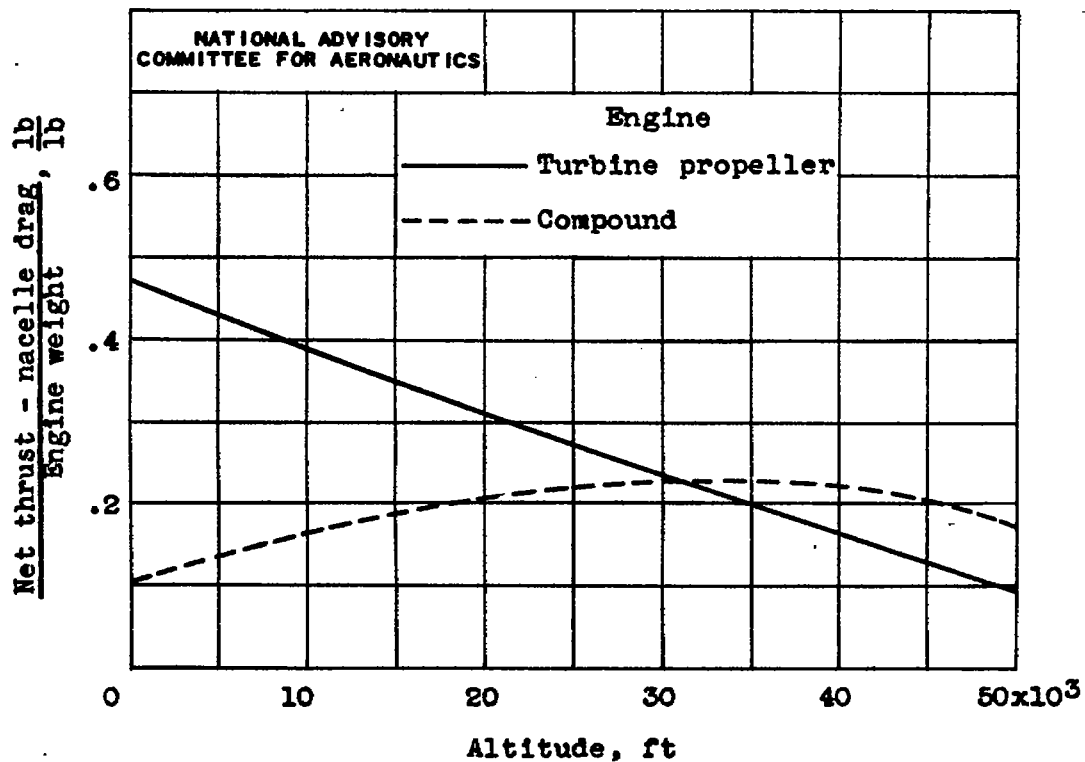Fig. II-6   NACA TN No. 1349

Figure II-6. - Comparison of net thrust minus nacelle drag per
   unit engine weight for compound and turbine-propeller engines.
   Flight speed, 500 miles per hour.

Fig. II-7

755-B
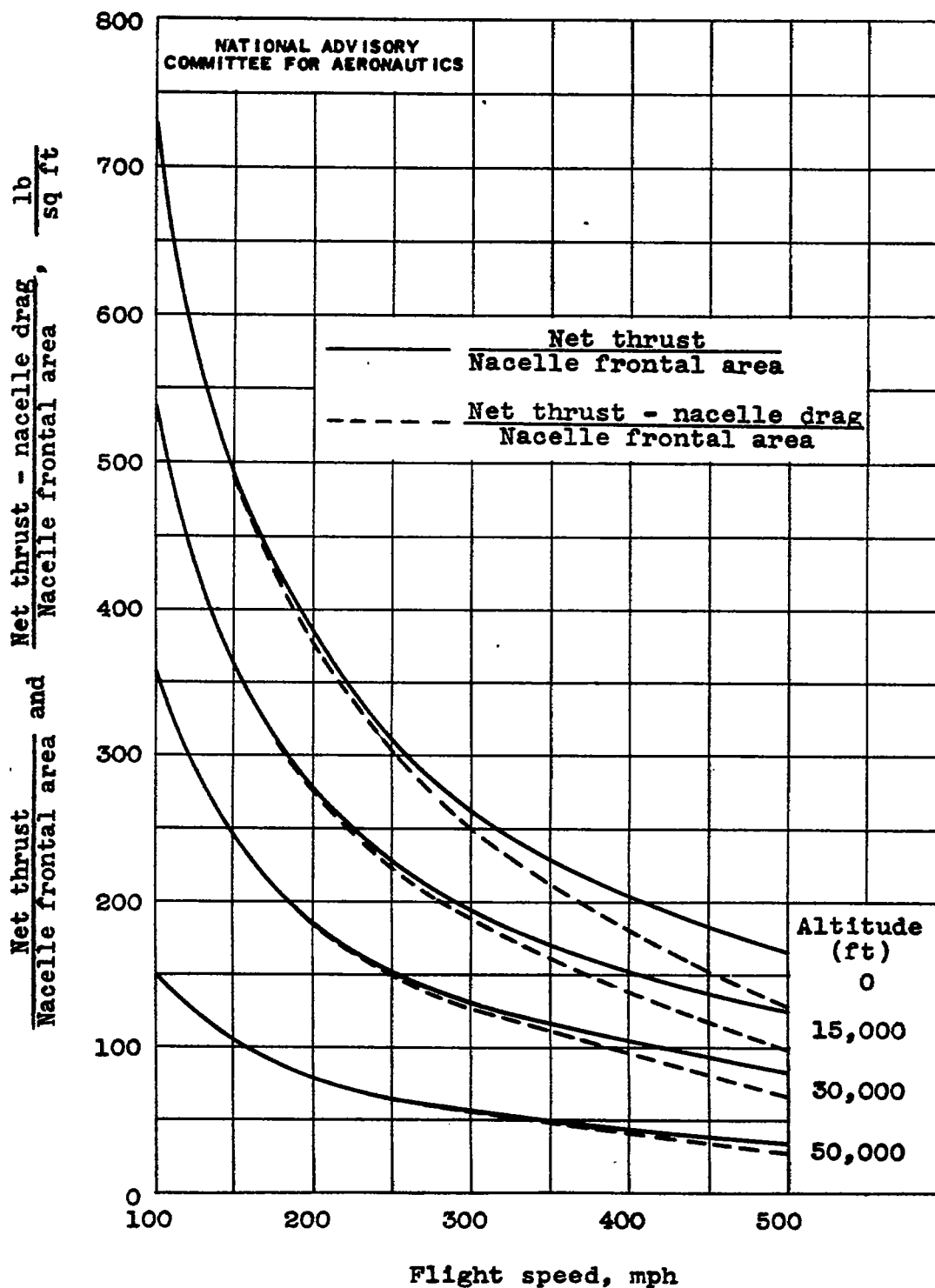
Figure II-7. - Variation of net thrust and net thrust minus nacelle drag per unit nacelle frontal area with flight speed and altitude for turbine-propeller engine.

Fig. II-8a

(a) Constant L/D, 18.
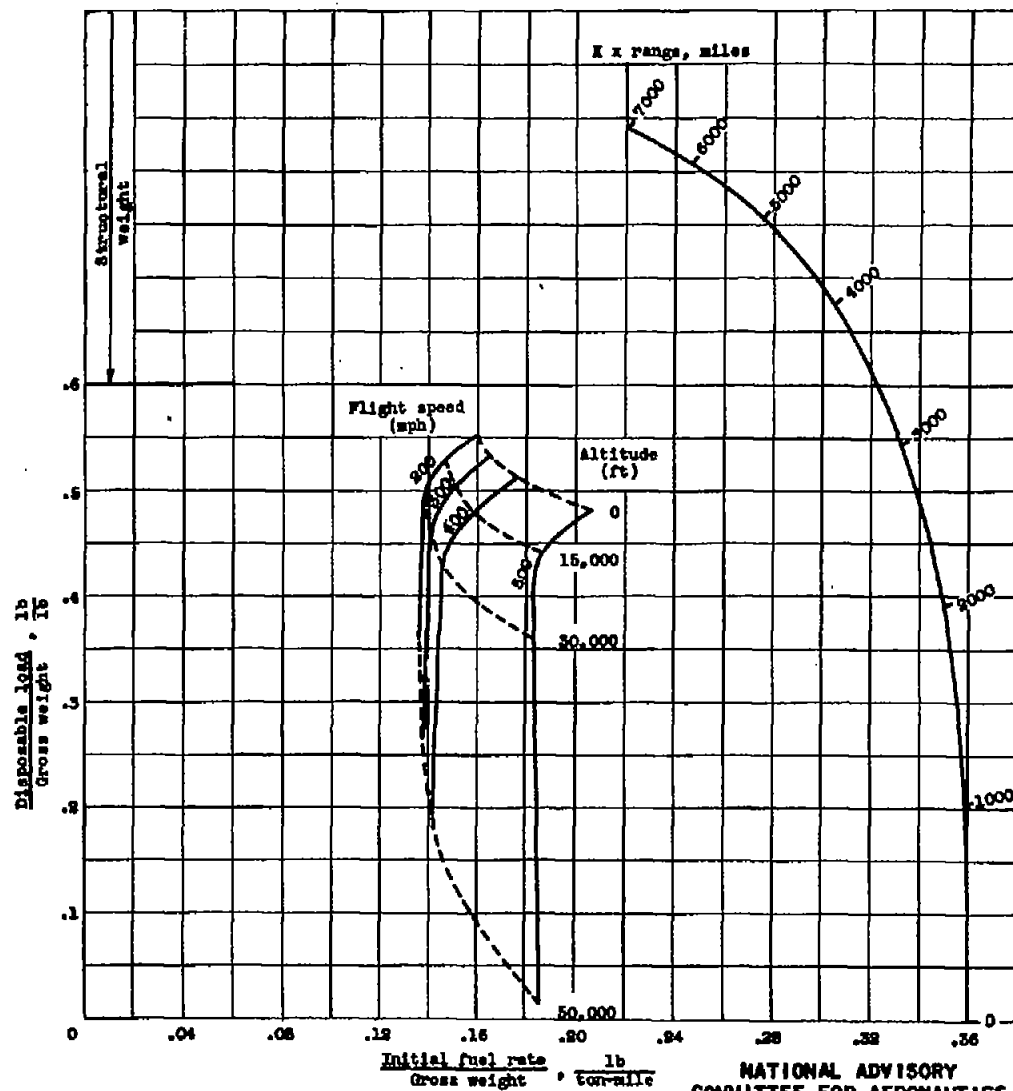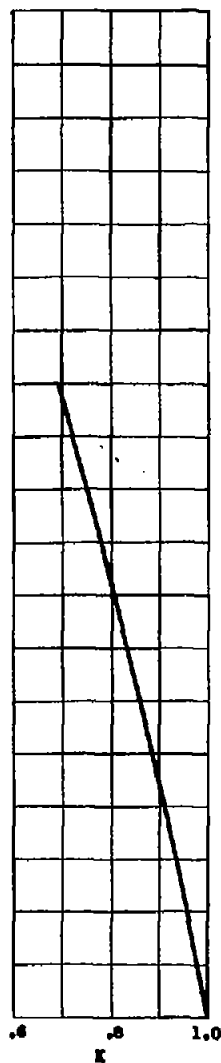
Figure II-8. - Load-range characteristics of turbine-propeller engine.

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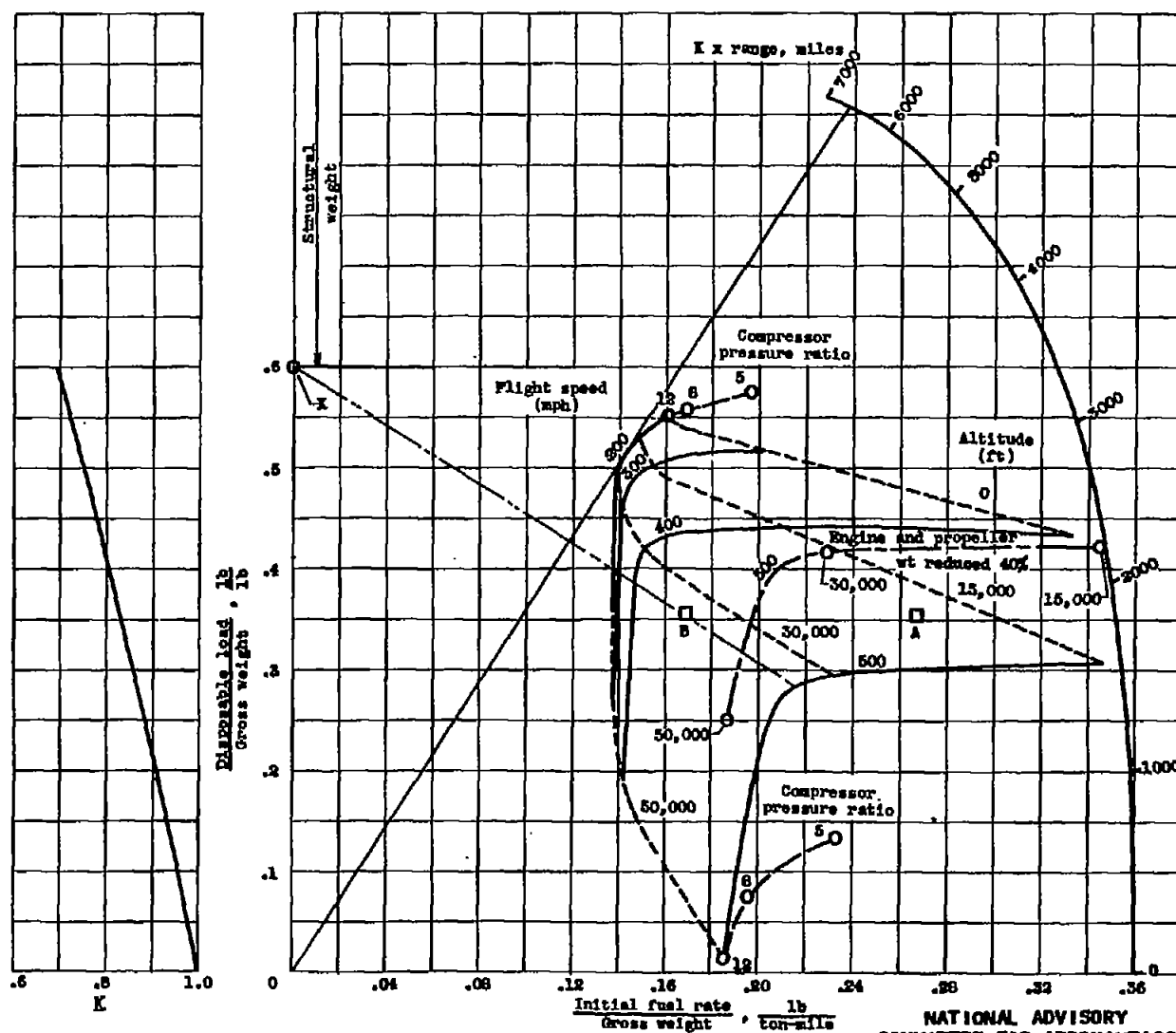
(b) Wing loading limited to 80 pounds per square foot.

Figure II-8. - Concluded. Load-range characteristics of turbine-propeller engine.

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Fig. II-8b

Fig. II-9

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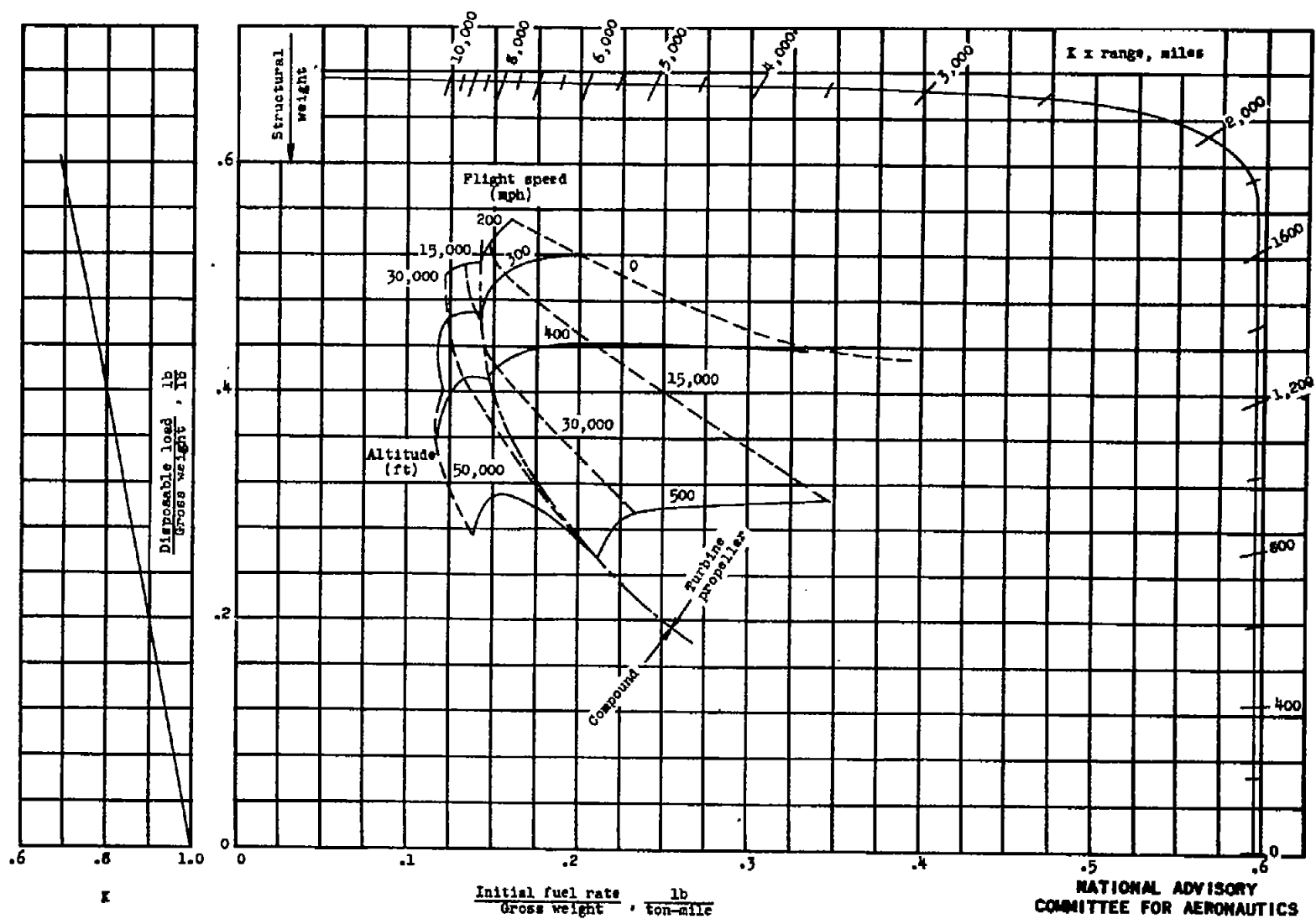
Figure II-9. — Comparison of load-range characteristics of compound and turbine-propeller engines. Wing loading limited to 50 pounds per square foot.

K x range, miles

5000

4000

Structural weight

.6

Engine weight (neglecting propeller weight)
Equivalent brake horsepower at static sea-level conditions

.5

.4

0.40

.60

.80

3000

1.00

2000

Disposable load , lb
Gross weight

.3

50,000

50,000

30,000

30,000

15,000

Altitude (ft)

50,000

.2

50,000

Turbine-propeller
engine

1000

.1

Compound
engine

0

.6 .8 1.0 | 0 .04 .08 .12 .16 .20 .24 .28 .32 .36 .40

K

$$\frac{\text{Initial fuel rate}}{\text{Gross weight}} , \frac{\text{lb}}{\text{ton-mile}}$$

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Figure II-10. - Effect of reduction in weight of turbine-propeller engine on disposable load and range at 500 miles per hour. Wing
loading limited to 80 pounds per square foot; pressure ratio, 12.

Fig. II-10

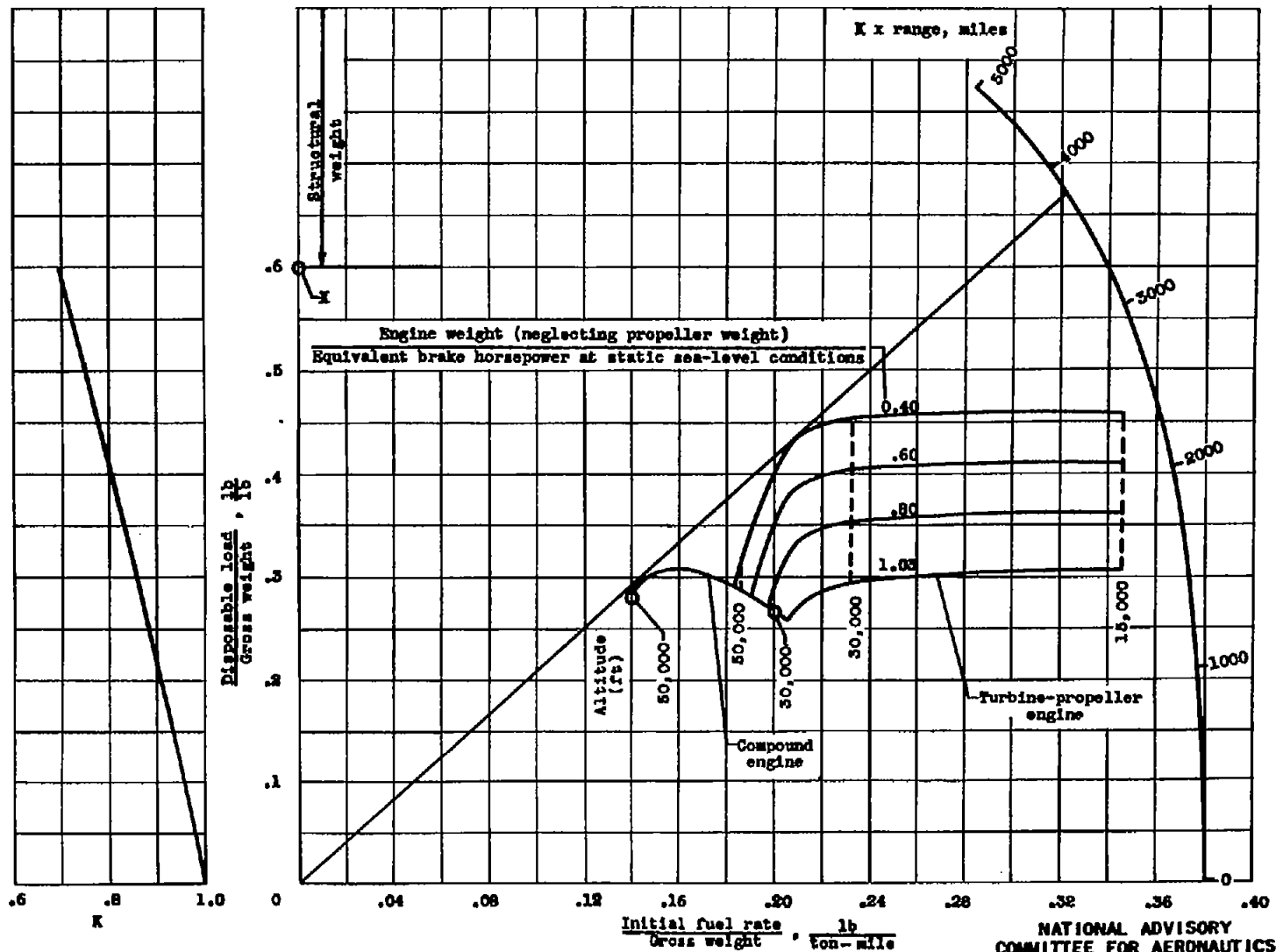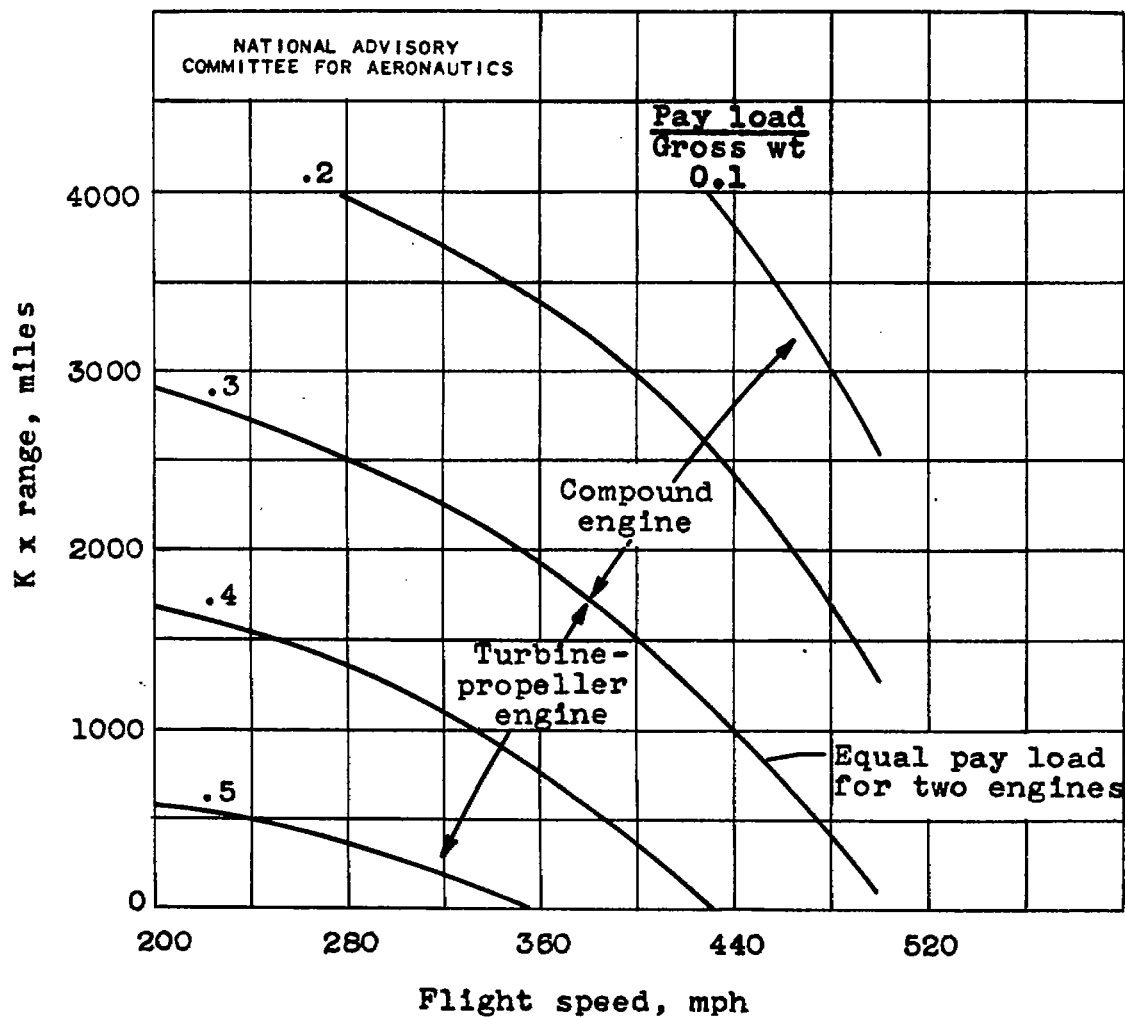Fig. II-11   NACA TN No. 1349

Figure II-11. - Speed-range spectrum showing zones where
compound and turbine-propeller engines have greater pay-
load carrying capacity.

III - THE TURBOJET ENGINE

Description

A turbojet engine produces a propulsive thrust by drawing in air, accelerating it to a high velocity, and discharging the high-velocity air in a rearward direction. Thrust results from the reaction of the acceleration of the air.

A schematic diagram of a turbojet engine is shown in figure III-1. Air is drawn in at the compressor inlet and is compressed to a high pressure in the compressor; the high-pressure air passes into a combustion chamber where fuel is added and burned and the high-temperature products of combustion expand through the turbine that drives the compressor; and finally, the high-energy gases leaving the turbine expand through a nozzle as a jet in the atmosphere.

Engine Performance

Fundamentally, both the turbojet and the propeller produce a propulsive thrust by accelerating air in a rearward direction. The turbojet differs from the propeller in that a large acceleration is given to a small mass of air; whereas the propeller gives a small acceleration to a large mass of air. In either case, the propulsive thrust equals the product of the mass of air handled and the increase in velocity of the air passing through the turbojet or propeller. The kinetic energy imparted to the air by the turbojet is greater than that imparted by the propeller because the kinetic energy equals the product of the mass of air and the square of the velocity; whereas the thrust is proportional to the first power of the velocity. In other words, the propulsive efficiency of a turbojet is much poorer than that of a propeller. The approximate propulsive efficiency of a turbojet-powered aircraft flying at 340 miles per hour at sea level is 37 percent; doubling the flight speed to 680 miles per hour raises the propulsive efficiency to 60 percent. In contrast to the low value of 37 percent at 340 miles per hour, propeller efficiencies of 85 percent are obtainable. At transonic and supersonic speeds the propeller efficiency decreases greatly because of compressibility effects. At the same time the propulsive efficiency of the turbojet continues to increase with increasing flight speed. It can therefore be concluded that at subsonic flight speeds a turbojet will always be handicapped by low propulsive efficiencies, but at supersonic speeds this handicap is overcome.

The over-all efficiency of a turbojet is a function not only of propulsive efficiency but of the thermal cycle efficiency. It is

well known that the efficiency of the ideal cycle increases with
increasing compressor pressure ratio and at first glance it would
appear that the highest possible compressor pressure ratio would be
desirable. Actually, because of losses in the compressor and the
turbine and because the available turbine materials limit the com-
bustion temperatures, there is a finite compression ratio at which
best economy is obtained. The compressor pressure ratio at which
best thrust is obtained from an engine with a given air capacity is
considerably lower than the compressor pressure ratio for best
economy. Most current turbojet engines operate with compressor
pressure ratios close to the value for maximum thrust.

The compressor pressure ratio at which best thrust is obtained
decreases with increasing flight speed and finally at a flight
speed between 1400 and 1500 miles per hour the optimum compressor
pressure ratio falls to a value of 1.0. At this value, there is
no compression in the compressor and the engine is operating essen-
tially as a low-temperature ram jet.

The thrusts that can be obtained from a series of engines,
each operating at the compressor pressure ratio for maximum thrust
at flight speeds between 0 and 1500 miles per hour and altitudes of
sea level, 30,000, and 50,000 feet, are shown in figure III-2.
These curves do not represent any single engine; instead, each
point on the curves represents a separate engine designed to operate
at the optimum compressor pressure ratio for the conditions of
altitude and speed indicated. The values shown in figure III-2
were estimated by assuming a compressor efficiency of 85 percent, a
turbine efficiency of 90 percent, and a combustion efficiency of
95 percent with a turbine-inlet temperature of 1540° F. The air-
handling capacity of the engine was assumed to be 13 pounds per
second per square foot of frontal area at sea level and zero flight
speed. At other flight conditions, the compressor-inlet Mach
number was assumed to be the same as the value corresponding to these
conditions.

In the range of subsonic speeds, flight speed has relatively
little effect upon the thrust; at supersonic flight speeds, how-
ever, the thrust significantly increases with increasing flight
speed. (See fig. III-2.) At a speed of 1500 miles per hour and
sea-level altitude, the thrust reaches 2000 pounds per square foot
and, in terms of horsepower, this thrust is equivalent to 8000 horse-
power per square foot of engine frontal area. Increasing the
altitude decreases the thrust because of the decreasing air density.

The drag of the engine nacelle becomes large at high flight speeds and, consequently, the net thrust available from the jet-engine installation is considerably less than the values shown in figure III-2. Values of engine thrust minus nacelle drag are shown in figure III-3. A comparison of this figure with figure III-2 shows the great reduction in available thrust at supersonic speeds.

The weight of a jet engine is, of course, also an important consideration. The thrust per unit engine weight based upon values given in figure III-2 is presented in figure III-4. The weights of the engines have been estimated from the weight of a standard turbojet. The weights of the compressor and the turbine were corrected by assuming that these weights are proportional to the logarithm of the pressure ratios; the weights of the other elements of the engine were not altered. Estimates based upon these assumptions resulted in a value of 2.62 pounds thrust per pound engine weight at sea level and zero flight speed (fig. III-4). Higher values for the ratio of thrust to engine weight actually have been obtained and future developments may result in additional increases.

The fuel economies, expressed as thrust specific fuel consumptions, are given in figure III-5 for conditions corresponding to the thrusts given in figure III-2. An increase in flight speed increases the thrust specific fuel consumption; from zero flight speed at sea level, the fuel consumption increases from 0.85 to a value of 1.9 pounds per hour per pound of thrust at 1400 miles per hour. An increase in altitude improves the fuel consumption because of the reduction in air temperature with increasing altitude.

The thrust and fuel consumption shown in figures III-2 and III-5, respectively, have been used to estimate the performance of the subsonic and supersonic airplanes powered by turbojet engines.

Load-Range Characteristics

Subsonic flight speeds. - The range of subsonic aircraft powered by turbojet engines is estimated using the same assumptions regarding the airplane characteristics as were used in the preceding parts of this report; these assumptions are presented in the appendix. Results of the calculations at a lift-drag ratio of 18 are shown in figure III-6(a). The most important result shown in this figure is the great reduction in the fuel rate per ton-mile with increasing flight speed. This result is directly contrary to the findings presented in the preceding parts of this report for the engines utilizing propellers wherein the fuel rate per ton-mile increased with

increasing flight speed. This decrease in fuel rate with increasing
flight speed is a direct reflection of the improvement in propulsive
efficiency of a turbojet engine with increasing flight speed. Best
economy and greatest range is seen to be obtained at the highest flight
speed considered, 550 miles per hour. Altitude has relatively small
effect upon the range at high flight speed. The range factor that is
found by drawing a line through the origin and tangent to the curve
representing 550 miles per hour is 4130 miles, the K factor for the
disposable load at the point of tangency is 0.72, and the range is
5740 miles.

Increasing the compression ratio to values above that required
for maximum power improves the fuel consumption but reduces engine
thrust. The effects of increasing the compression ratio upon range
at 550 miles per hour and an altitude of 30,000 feet are also shown
in figure III-6(a). Increasing the compression ratio from the value
for maximum thrust 7.8, to the value for best economy 18, reduces
the fuel rate per ton-mile without seriously affecting the disposable
load and, consequently the range is improved. The range factor
(K × range) at a compression ratio of 18, flight speed of 550 miles
per hour, and altitude of 30,000 feet is 4700 miles, the value of
K is 0.735, and the range is therefore 6400 miles.

As was previously mentioned, the results shown in figure III-6(a)
apply to the airplane operating at the maximum lift-drag ratio of 18
at all flight speeds and altitudes. This assumption results in extremely
high wing loadings at high flight speeds and particularly at low altitudes.
These high wing loadings make it necessary to use special methods for
launching or assisting in take-off of the aircraft.

The curves shown in figure III-6(b) were estimated by selecting
a lift-drag ratio to give a wing loading of 80 pounds per square foot
except in cases where a lift-drag ratio of 18 gives wing loadings less
than 80 pounds per square foot. In such cases the lift-drag ratio was
assumed to be 18.

At low flight speeds or at high altitudes, the wing loading at a
lift-drag ratio of 18 is less than 80 pounds per square foot; consequently,
the values of disposable load and fuel consumption per mile are the same
as those shown in figure III-6(a). At these flight conditions, the fuel
rate per ton-mile decreases with increasing flight speed, as has been
previously discussed. At speeds somewhat above the limiting speed at
which the wing loading equals 80 pounds per square foot, the fuel rate
per ton-mile increases with increasing flight speed because the reduction
in aerodynamic efficiency accompanying the reduction in lift-drag ratio
more than counteracts the improvement in propulsive efficiency with
flight speed. As an example, figure III-6(b) shows that at an altitude of

30,000 feet, the fuel rate per ton-mile decreases with increasing flight speed up to a flight speed of about 400 miles per hour beyond which the fuel rate per ton-mile increases with further increase in flight speed.

The best range of 4670 miles was estimated from figure III-6(b). Flight at substantially higher speeds than 550 miles per hour will not improve range because compressibility effects will increase drag and reduce the lift-drag ratio. Also flight at high altitudes will not improve range because the reduction of thrust with altitude reduces the disposable load as can be seen in figure III-6(b). Flight at high speed and low altitude results in extremely poor fuel economy and range. In particular, at sea level and 550 miles per hour the range is reduced to 1410 miles and the fuel consumption is about four times greater than that obtained at the most economical speed and altitude.

A comparison of the performance of airplanes powered by compound, turbine-propeller, and turbojet engines is shown in figure III-7. These curves represent performance in cases where the wing loading is limited to 80 pounds per square foot. The best range of the turbojet engine is much less than the best range of either the compound or the turbine-propeller engine. If a flight speed of 550 miles per hour is desired, the range of the turbojet exceeds the range of the other two engines.

Supersonic flight speeds. - At supersonic flight speeds, the range estimates required an entirely different set of assumptions from those used at subsonic speeds. For these conditions the following assumptions were made: (1) The lift-drag ratio of the wing is assumed to be 7 instead of the previous value of 18 for the entire airplane less nacelles; (2) the size of the fuselage required to accommodate the disposable load was estimated and the drag of the fuselage at each flight speed and altitude was calculated; (3) drag coefficient and diffuser efficiencies were selected after a study of available data and the values of these coefficients and efficiencies are given in the appendix; (4) the weight of the structure is 0.3 of the gross weight; and (5) the tank weight is 10 percent of the fuel weight.

The gross weight W_g of the airplane is given by

$$W_g = (F - D_n - D_f) \frac{L}{D} \tag{1}$$

where

F net thrust of engine, pounds

D_n nacelle drag, pounds

D_f fuselage drag, pounds

L/D lift-drag ratio of wing

The total disposable load W_d is

$$W_d = (1 - a) W_g - W_e \qquad (2)$$

where

a ratio of structure weight to gross weight

W_e engine weight, pounds

Fuselage size was estimated on the assumption that the density of the disposable load was 50 pounds per cubic foot. The fuselage drag D_f equals the sum of the skin-friction drag and the wave drag. For a fuselage with a length-diameter ratio of 12 and with conical ends having cone angles of 20°, the drag was calculated from the equation:

$$D_f = q_0 \left(\frac{W_d}{\rho_f} + \frac{W_d + W_e}{700} \right)^{2/3} (0.4528 \, C_{D,I} + 8.34 \, C_{D,F}) \qquad (3)$$

where

q_0 dynamic pressure (incompressible), pounds per square foot

ρ_f fuel density

$C_{D,I}$ wave-drag coefficient

$C_{D,F}$ skin-friction drag coefficient, 0.003

Values of $C_{D,I}$ are given in the appendix. The term $\frac{W_d}{\rho_f}$ is volume of fuel; $\frac{W_d + W_e}{700}$ is the volume allowed for controls, which is based upon 2 cubic feet per ton of gross weight.

Equations (1), (2), and (3) were simultaneously solved to obtain W_d and W_g.

Unlike the subsonic case, the results are not independent of the size of engine chosen because the drag of the fuselage

increases with the square of a linear dimension of the fuselage; whereas, the load-carrying capacity increases with the cube of a linear dimension. Consequently, the fuselage drag per pound of disposable load is less for a large airplane than for a small one.

In order to permit comparisons of the performance of airplanes powered by turbojet, turbo-ram-jet, and ram-jet engines, the frontal areas of all turbojets were fixed at 12.5 square feet. The resultant gross weights of airplanes designed to fly at 12 flight conditions are given in the following table:

Altitude (ft) Flight speed (mph)	0	30,000	50,000
	Gross weight (lb)		
900	10,900	14,000	8,730
1100	11,300	16,800	10,800
1300	12,900	21,200	14,100
1400	14,300	24,100	16,400

Results of the calculations for supersonic flight are shown in figure III-8. A graph of the K factor is not shown because at supersonic speeds the parasitic drag of the nacelle and the fuselage is large compared to the drag of the wing and only a very small reduction in drag accompanies reduction of fuel load with flight duration. As an approximation, the value of K can therefore be assumed equal to 1.

It is immediately evident from figure III-8 that flight at low altitudes results in poor fuel economy and range and that flight at the highest altitude considered results in best economy. The best range is obtained at an altitude of 50,000 feet and 1400 miles per hour at which speed the compressor pressure ratio of the engine has dropped to almost 1 and the engine is operating essentially as a ram jet. The value of the maximum range is 1330 miles. Even greater range would be obtained at higher altitudes.

At the best range condition shown in figure III-8, that is, 1400 miles per hour at 50,000 feet, the gross weight of the airplane corresponding to the point shown is 16,400 pounds, as given in the foregoing table. The effect of gross weight on performance

is illustrated by a computation of the performance at 1400 miles per hour and 50,000 feet for gross weights of 8200 pounds and 49,200 pounds. These points are included in figure III-8.

At supersonic speeds, increasing the compression ratio to values greater than that required to give best thrust results in less range.

Conclusions

It can be concluded from this study that at flight speeds less than 550 miles per hour the best range of a turbojet-powered air-plane is considerably less than the best range of airplanes powered by a compound or a turbine-propeller engine. At flight speeds above 550 miles per hour, however, the range of the turbojet-powered airplane is greater than the range of an airplane powered by a compound or a turbine-propeller engine. The best range of the turbojet-powered airplane with a wing loading limited to maximum value of 80 pounds per square foot is obtained at maximum altitude and maximum flight speed. The best range of supersonic aircraft equipped with turbojet engines investigated in this study was obtained at the maximum altitude and engine speed considered (namely, 50,000 ft and 1400 mph). The best supersonic range found in these calcula-tions was roughly one-fourth of the range obtainable by subsonic aircraft powered by turbojet engines.
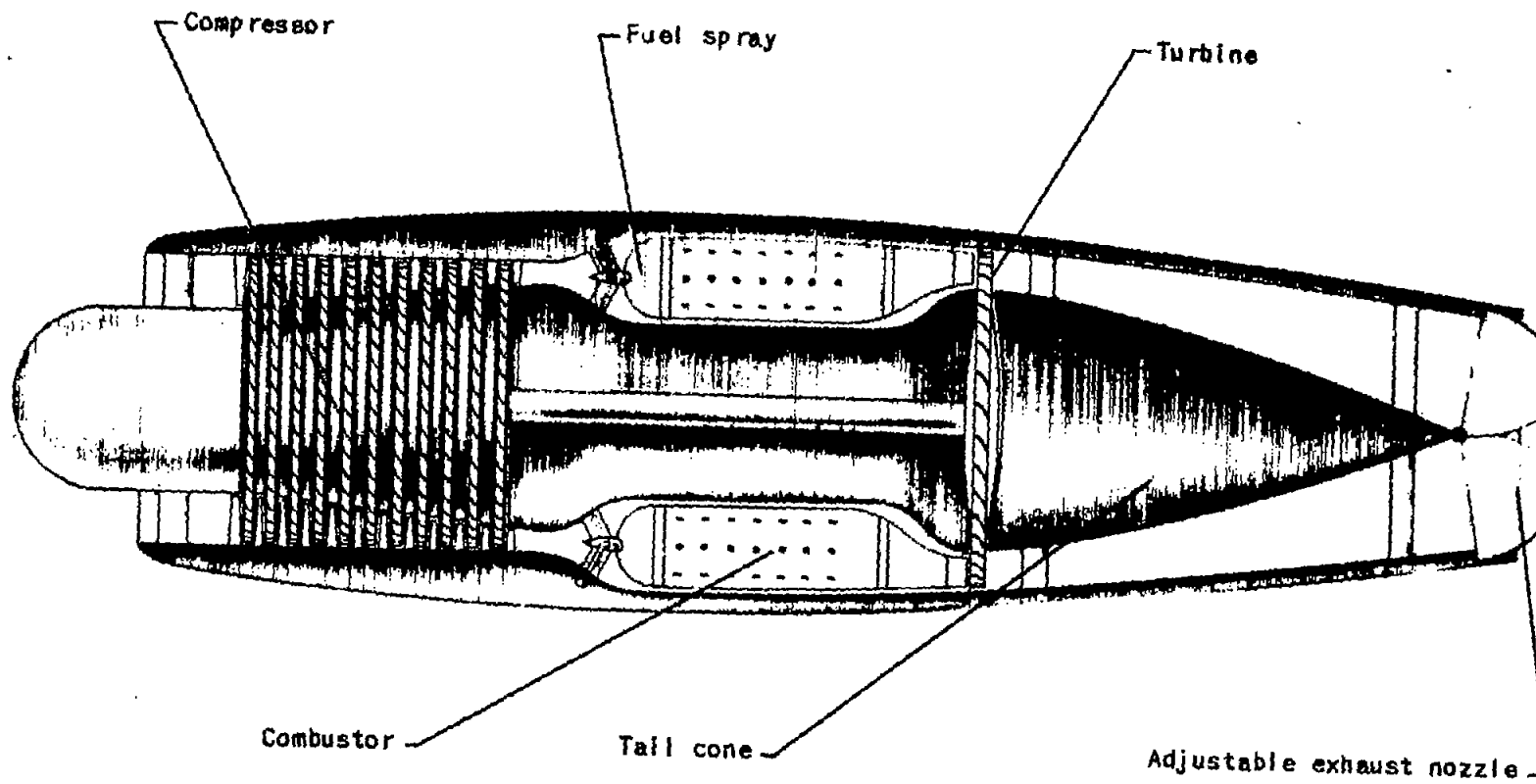
Figure III-1. - Diagrammatic sketch of turbojet engine.
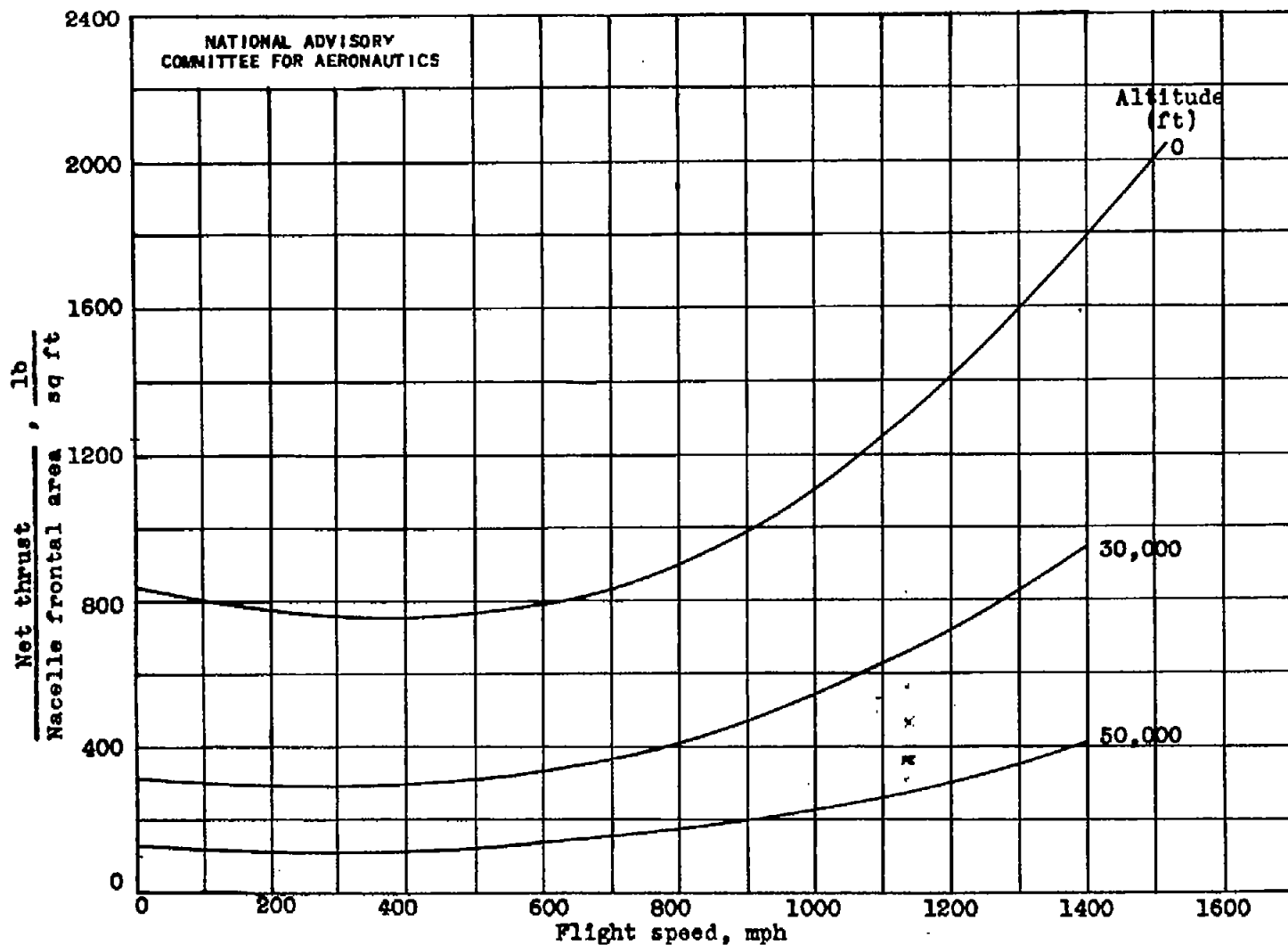
Fig. III-1

Fig. III-2

Figure III-2. — Variation of net thrust per unit nacelle frontal area with flight speed and altitude for turbojet engine. Engine operating at compressor pressure ratio for maximum thrust.
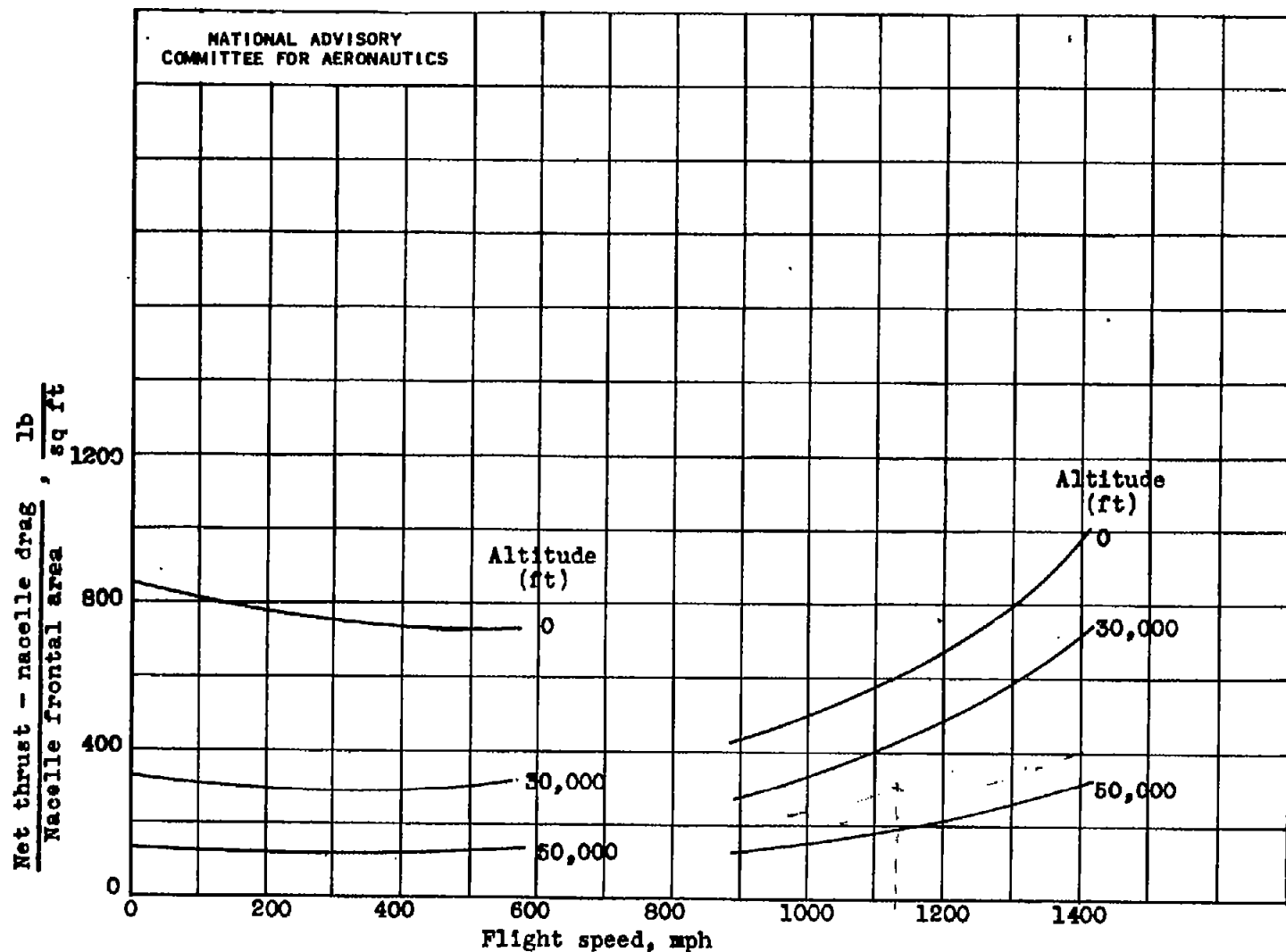
Figure III-3. – Variation of net thrust minus nacelle drag per unit nacelle frontal area with flight speed and altitude for turbojet engine. Engine operating at compressor pressure ratio for maximum thrust.

Fig. III-3

Fig. III-4

$$\frac{\text{Net thrust}}{\text{Engine weight}} , \frac{\text{lb}}{\text{lb}}$$

Flight speed, mph

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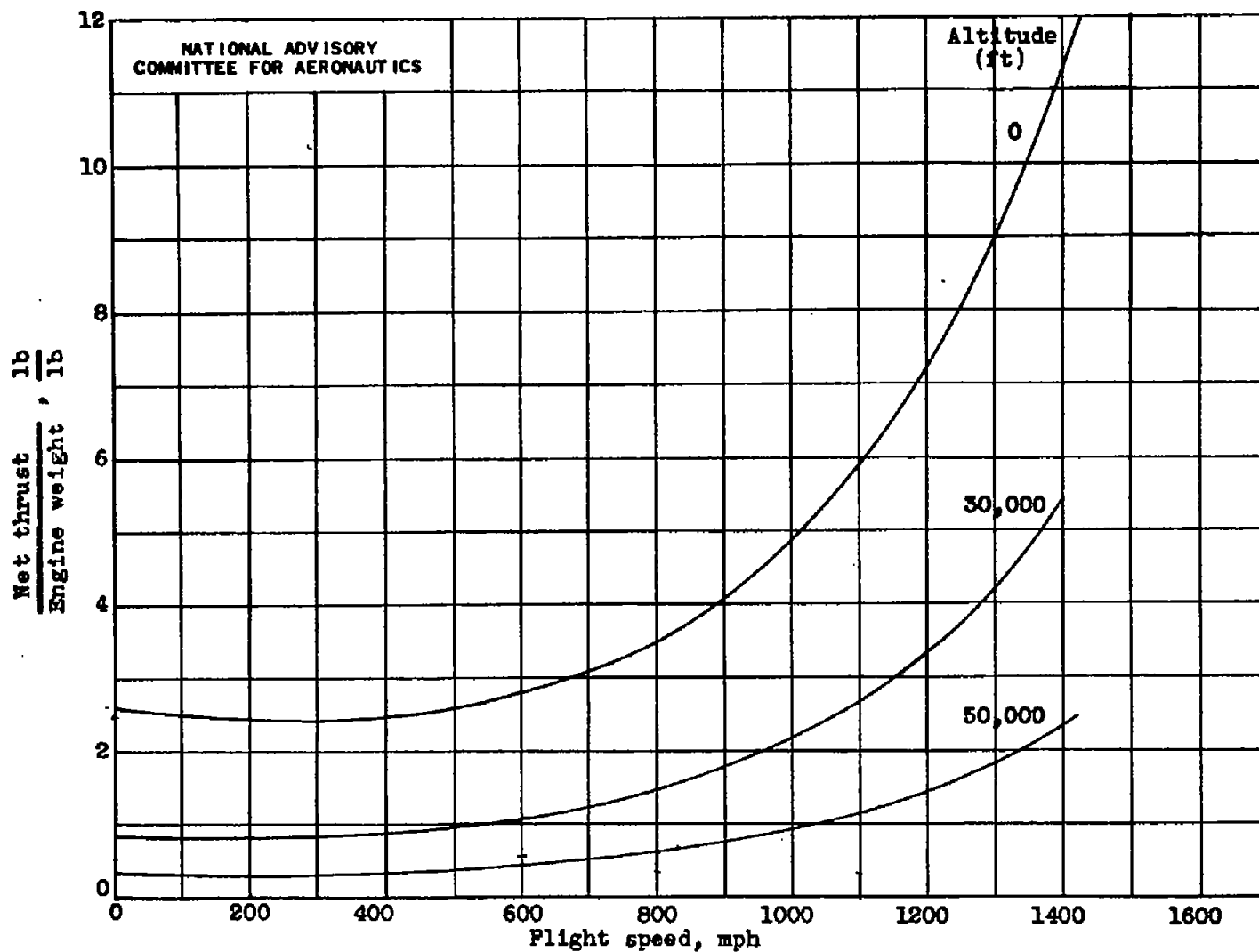
Altitude
(ft)

0

30,000

50,000

Figure III-4. – Variation of net thrust per unit engine weight with flight speed and altitude
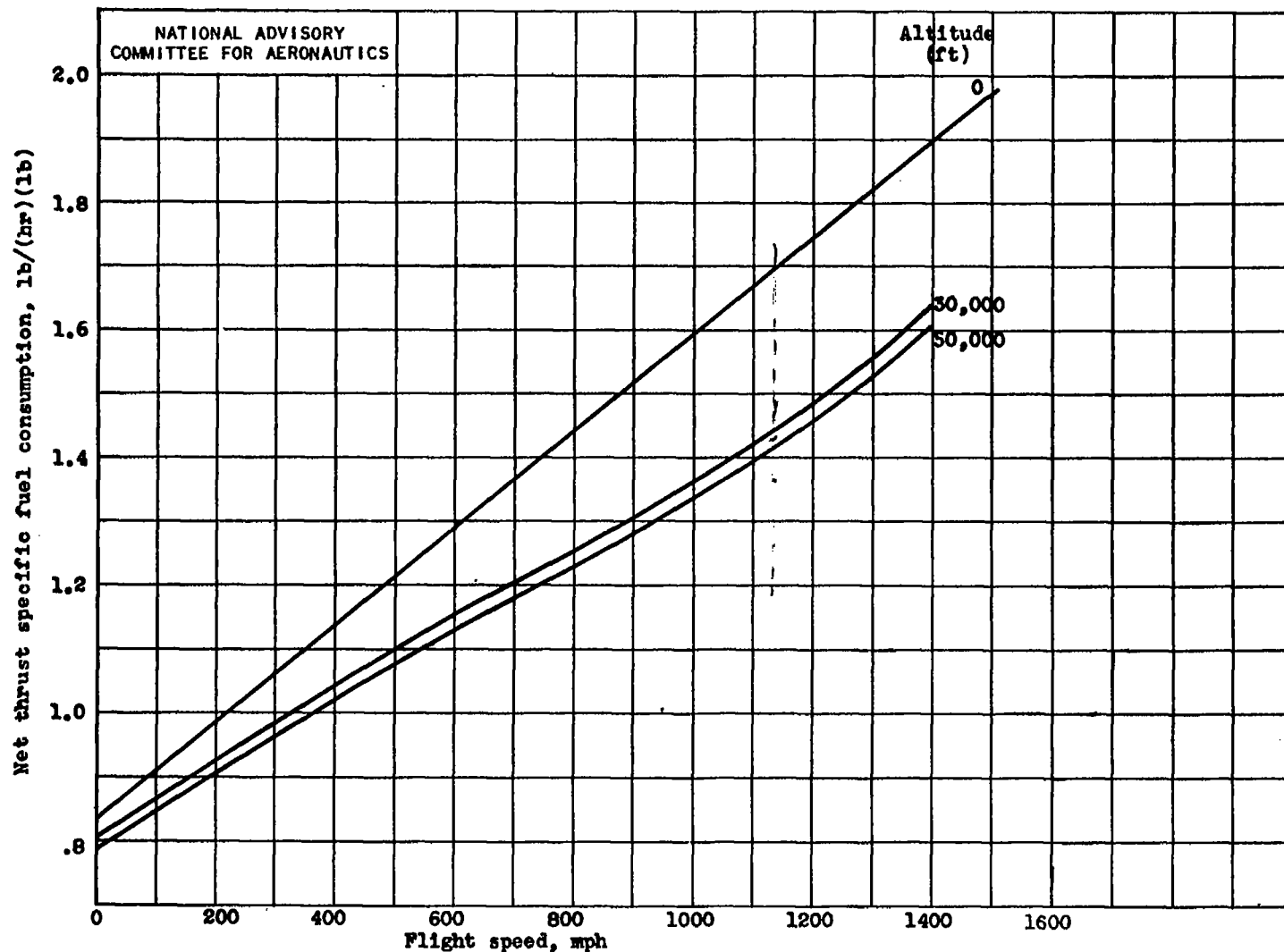for turbojet engine. Engine operating at compressor pressure ratio for maximum thrust.

Figure III-5. — Variation of net thrust specific fuel consumption with flight speed and altitude for turbojet engine. Engine operating at compressor pressure ratio for maximum thrust.

Fig. III-5

Fig. III-6a

(a) Constant L/D, 18.

Figure III-6. - Load-range characteristics of turbojet engine at subsonic flight speeds. Engine operating at compressor pressure ratio for maximum thrust.

(b) Wing loading limited to 80 pounds per square foot.

Figure III-6. - Concluded. Load-range characteristics of turbojet engine at subsonic flight speeds. Engine operating at compressor pressure ratio for maximum thrust.

Fig. III-6b

Fig. III-7

Figure III-7. - Comparison of load-range characteristics of compound, turbine-propeller, and turbojet engines for subsonic flight speeds. Wing loading limited to 50 pounds per square foot.
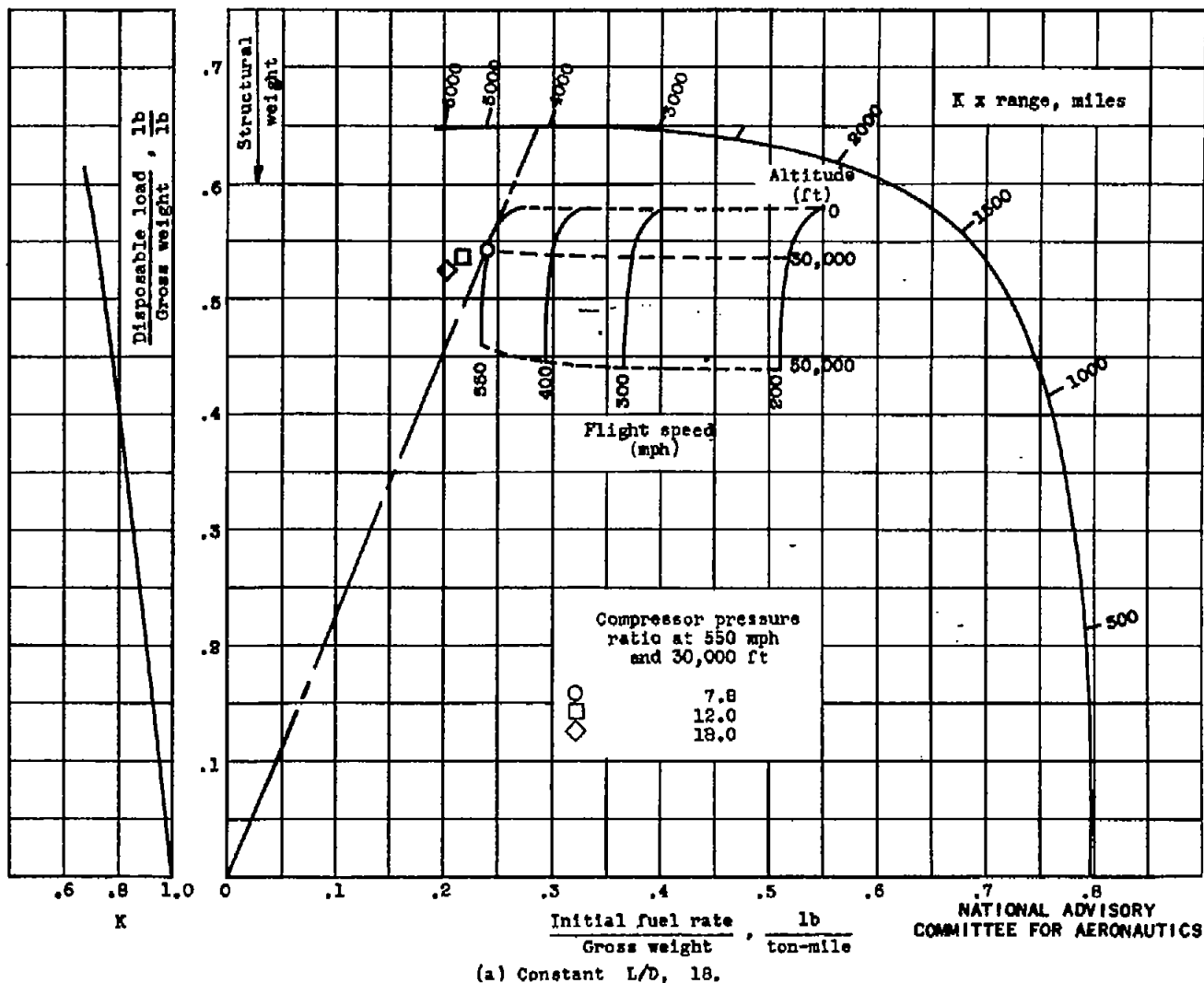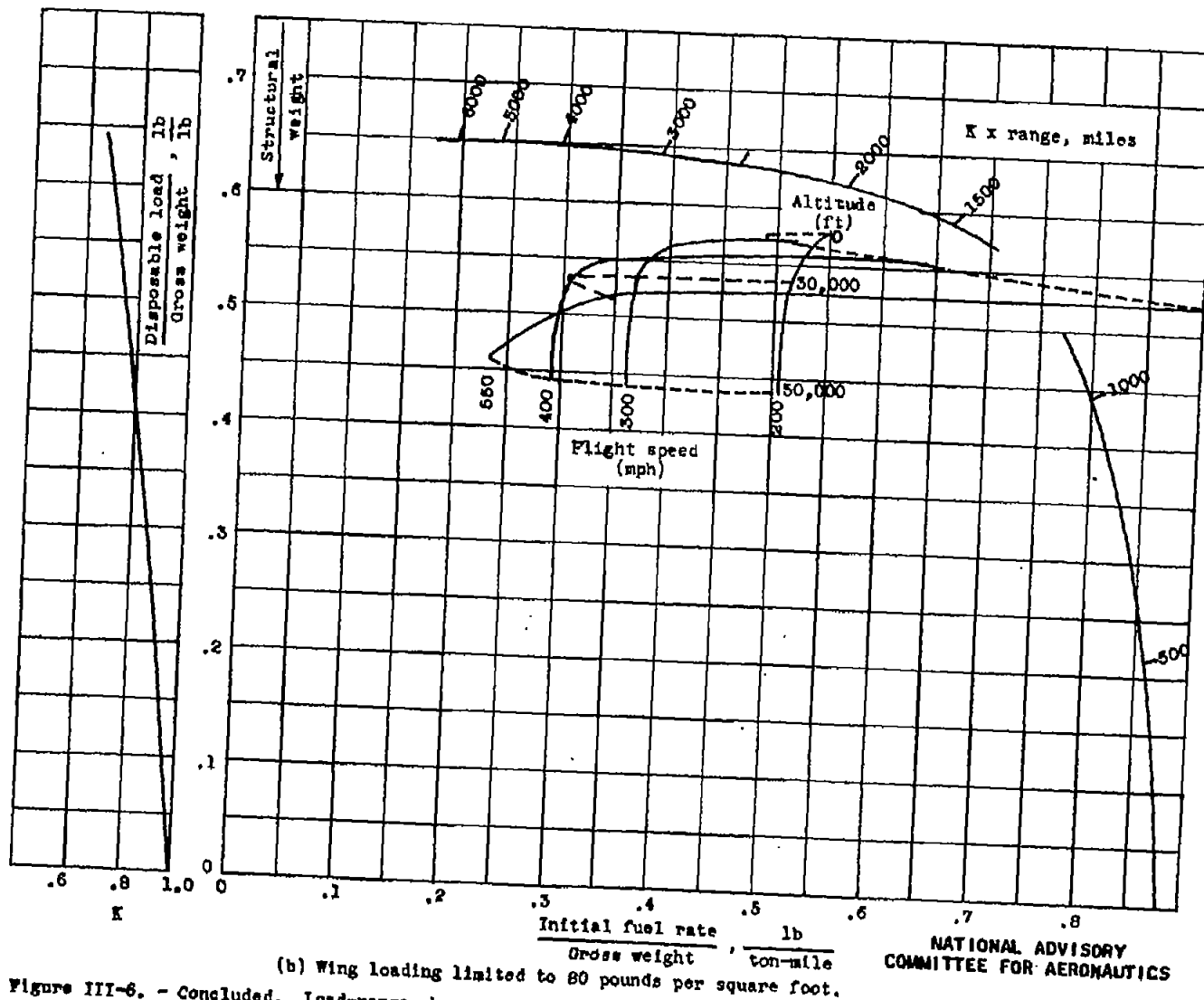
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Figure III-8. — Load-range characteristics of turbojet engine at supersonic flight speeds. Engine operating at compressor pressure ratio for maximum thrust; constant L/D for wing, 7.
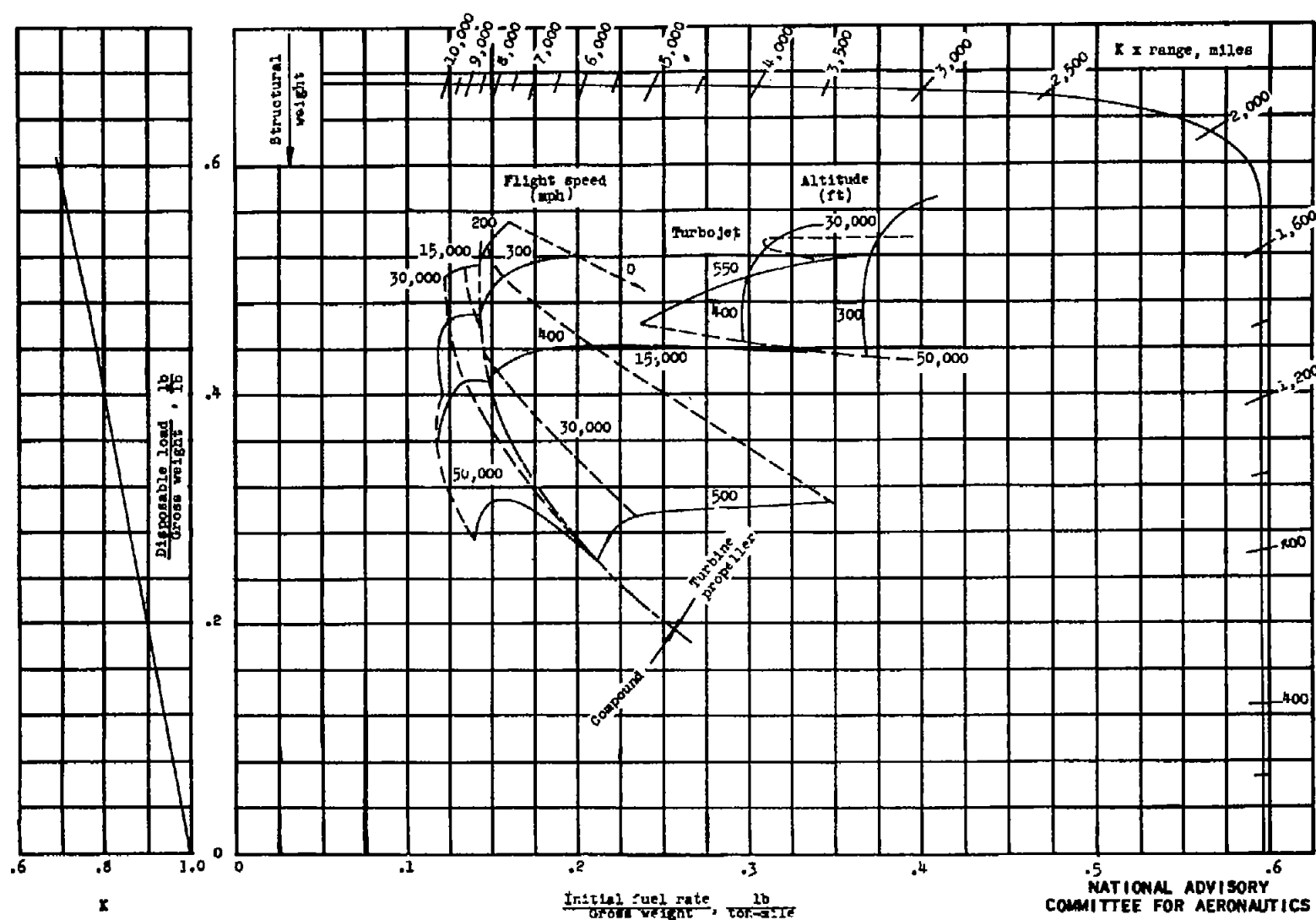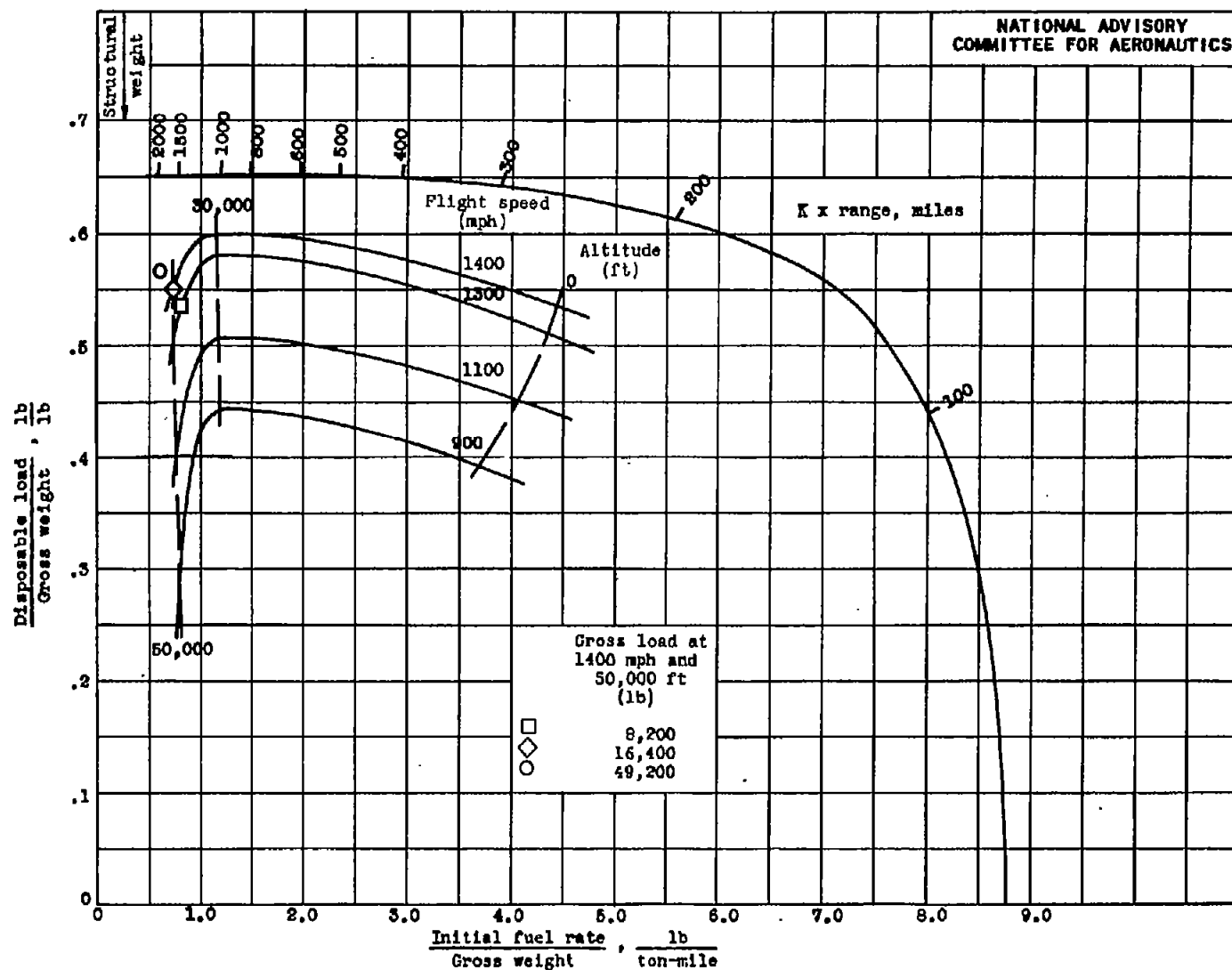
IV - THE TURBO-RAM-JET ENGINE

Description

The turbo-ram-jet engine is diagrammatically illustrated in figure IV-1. This engine consists essentially of a conventional turbojet engine with provision for reheating the gas between the turbine discharge and the exhaust nozzle. In this manner, it is possible to obtain higher gas temperatures in the exhaust jet than can be withstood by the turbine. As its name implies, the turbo-ram-jet engine may be considered as a combination of a turbojet engine and a ram-jet engine in which the inlet conditions are equal to the turbine-discharge conditions of the turbojet. The cycle on which this engine operates is called tail-pipe burning or afterburning.

In this type of engine, it is necessary to reduce the gas velocities in the tail pipe below the values usually employed in turbojet engines to prevent the pressure drop in the tail pipe, caused by both the burning of the fuel and the drag of the necessary burner parts, from becoming excessive. The engine is therefore provided with a diffuser between the turbine discharge and the tail-pipe-burner inlet. An adjustable-area exhaust nozzle is also required to permit the engine to operate at rated turbine-inlet temperature over a range of exhaust-gas temperatures.

Engine Performance

In addition to the factors that affect the performance of turbojet engines, the principal parameters determining the performance of the turbo-ram-jet engine are the temperature rise and the velocity of the gases in the tail pipe. Their effect is illustrated in figure IV-2 in which net thrust per unit nacelle frontal area is plotted against the exhaust-gas temperature for various values of the gas velocity at the tail-pipe-burner inlet. These curves are based on the performance of an engine fitted with a tail-pipe burner having a total-pressure drop due to friction of 0.4 times the dynamic pressure at the burner inlet and a turbine-discharge diffuser efficiency of 75 percent. The calculations are also based on flight conditions of 500 miles per hour at sea level although the same general trends would be obtained at any other flight condition.

When the velocity in the tail pipe is high, a sonic limit is reached beyond which it is impossible to add heat to the gases and still maintain constant engine conditions. The limiting temperature for the lower gas velocities is obtained when the over-all fuel-air ratio is stoichiometric (0.067). The rate of increase in

engine thrust with gas temperature is greatest when the gas velocity
is low because of the attendant lower momentum-pressure drop
(fig. IV-2). The importance of the pressure drop in the burner is
evident from the considerable gains in thrust which may be realized
by reducing the velocity. For all subsequent calculations, the
diameter of the tail-pipe was assumed equal to the diameter of the
engine, which provided a burner-inlet gas velocity of 100 to 400 feet
per second depending on the flight speed and the altitude.

The variation of net thrust per unit nacelle frontal area with
flight speed at altitudes of sea level, 30,000, and 50,000 feet is
shown in figure IV-3 for exhaust-gas temperatures corresponding to
stoichiometric fuel-air ratio. The same component efficiencies and
the air-handling capacity were assumed for this engine as for the
turbojet engine discussed in part III of this report and the com-
pressor pressure ratio that provided maximum thrust was used. Each
point on the curves therefore corresponds to a different size engine.
Because the exhaust-gas temperature is approximately constant, this
optimum pressure ratio is obtained when the turbine-discharge pres-
sure is at maximum. The optimum pressure ratio for this engine is
considerably higher than for the turbojet engine and its variation
with flight speed and altitude is presented in the upper part of
figure IV-3. For zero flight speed at sea level, the optimum
pressure ratio is about 12 and decreases with increased flight
speed to a value of 1 at approximately 1800 miles per hour. At
an altitude of 50,000 feet, the optimum pressure ratio is about
twice that at sea level. The temperature at the tail-pipe-burner
outlet was obtained from the thermodynamic charts of reference 1
and both friction- and momentum-pressure losses in the tail pipe
were included in the calculations. A completely expanding exhaust
nozzle was assumed for all conditions that required an exit area
equal to or less than the engine frontal area. Where an exit area
greater than the nacelle frontal area was required for complete
expansion, a nozzle having an exit area equal to the nacelle
frontal area was used.

The net thrust increases rapidly with flight speed, par-
ticularly in the high-speed range, and decreases as the altitude
is increased (fig. IV-3). The flight speed at which the optimum
pressure ratio becomes equal to 1.0 is indicated by the dashed
limit line. At this point, the turbo-ram-jet engine is obviously
equivalent to a ram-jet engine. The net thrust of the turbo-ram-
jet engine, for stoichiometric fuel-air ratio, is from 100 to
200 percent greater than that of the turbojet engine, the differ-
ence increasing with increased flight speed, and reaches a value
of about 10,000 pounds per square foot of nacelle frontal area at

a speed of 1800 miles per hour at sea level. At an altitude of 50,000 feet, the net thrust is about 20 percent of the thrust produced at sea level.

The net thrust per unit nacelle frontal area for an over-all fuel-air ratio of 0.045 is shown in figure IV-4. This over-all fuel-air ratio was found to provide the greatest range for all flight conditions presented. The values of net thrust obtained for this fuel-air ratio are from 80 to 87 percent of the values shown in figure IV-3 for stoichiometric fuel-air ratio. The net thrust minus the drag of the engine nacelle is shown by the dashed lines in figure IV-4. At a flight speed of 1800 miles per hour at sea level, the engine produces a thrust of about 7000 pounds per square foot of nacelle frontal area after the nacelle drag has been subtracted from the engine net thrust. This value of thrust is reduced to approximately 1600 pounds per square foot of nacelle frontal area when the altitude is increased to 50,000 feet.

The net thrust specific fuel consumption is shown in figure IV-5 for the same range of flight speeds and altitudes. A combustion efficiency of 95 percent was assumed for the primary combustion and 90 percent for the secondary or tail-pipe combustion. Values of specific fuel consumption are shown for effective fuel-air ratios of stoichiometric and 0.045. These effective fuel-air ratios represent the amount of fuel that is burned in the engine; the actual fuel-air ratios are higher than these effective values because of the combustion inefficiency.

For both fuel-air ratios, the specific fuel consumption increases with flight speed at all altitudes and decreases as the altitude is increased. At an altitude of 50,000 feet, the specific fuel consumption for a fuel-air ratio of 0.045 increases from about 1.6 pounds per hour per pound of net thrust at very low flight speeds to about 2.2 at 1800 miles per hour. Based on thrust horsepower, the specific fuel consumption reaches a minimum value of about 0.45 pound per thrust horsepower-hour at a speed of 1800 miles per hour and an altitude of 50,000 feet. A comparison with the turbojet engine shows that the specific fuel consumption of the turbo-ram-jet engine is roughly twice that of the turbojet engine at low flight speeds and about $1\frac{1}{4}$ times as large at high flight speeds.

The net thrust per unit engine weight is shown in figure IV-6 for the same range of flight conditions and for an over-all fuel-air ratio of 0.045. The weight of the turbo-ram-jet engine was based on the weight of current turbojet engines with adjustments for changes in weight with compressor pressure ratio and plus the

estimated weight of the tail-pipe burner. The weight adjustments
for compressor pressure ratio were based on varying the weight of
the compressor and the turbine in proportion to the logarithm of
the pressure ratio. For flight conditions that resulted in high
internal pressures, sufficient additional weight for an engine cas-
ing and tail pipe was included to provide satisfactory hoop stresses.
The net thrust per unit engine weight increases rapidly with flight
speed, particularly at the higher altitudes, because of the simul-
taneous increase in engine thrust and decrease in engine weight as
the compressor pressure ratio is decreased. At static sea-level
conditions, the turbo-ram-jet engine delivers approximately 3 pounds
of thrust per pound of engine weight, which increases to approxi-
mately 23 at 1800 miles per hour.

Load-Range Characteristics

Subsonic flight speeds. - In figure IV-7(a), the disposable
load per airplane gross weight is plotted against the fuel consump-
tion per gross weight for subsonic flight speeds. An airplane lift-
drag ratio of 18 was used for the computations of these data. The
engine thrust and specific fuel consumption for both this and all
subsequent figures were obtained from figures IV-4 and IV-5, respec-
tively, that is, for an over-all fuel-air ratio of 0.045. The dis-
posable load per gross weight decreases with increased altitude and
is nearly independent of flight speed. The fuel rate per gross
airplane weight, however, decreases rapidly with increased flight
speed and decreases slightly with increased altitude. A maximum
value of the factor K × range of about 2750 miles is indicated
for the airplane characteristics assumed for this analysis. After
application of the K factor, indicated on the left side of the
figure, an actual range of about 3800 miles is obtained. This maxi-
mum range is obtained at a flight speed of 550 miles per hour and
an altitude of slightly over 30,000 feet.

If the wing loading is limited to a maximum of 80 pounds per
square foot, the load-carrying capacity and rate of fuel consump-
tion per gross weight shown in figure IV-7(b) is obtained. For
low-altitude and high-speed flight conditions, the load-carrying
capacity is slightly reduced from the values obtained at maximum
lift-drag ratio and the fuel consumption is greatly increased.
Thus, whereas the maximum range is nearly independent of altitude
for maximum lift-drag ratio, the advantages of high-altitude flight
are clearly evident when the wing loading is fixed. For example,
at a speed of 550 miles per hour, the K × range is increased from
660 to 2700 miles as the altitude is increased from sea level to
50,000 feet.

The airplane flight characteristics shown in figure IV-7(b) are reproduced in figure IV-8 together with the corresponding plots for the compound, the turbine-propeller, and the turbojet engines. The turbo-ram-jet engine provides a slightly greater disposable load than the turbojet engine at the expense of a greatly increased fuel rate. The maximum range for the turbo-ram-jet engine is about 75 percent as large as for the turbojet engine.

The principal field of application of the turbo-ram-jet engine at subsonic speeds is therefore as a short-duration thrust-augmentation device. By merely shutting off the fuel flow to the tail-pipe burner and reducing the exhaust-nozzle area, this engine becomes essentially a turbojet engine. By this means, the inherent high thrust of the turbo-ram-jet engine may be used for take-off and climb and the lower fuel-consumption characteristics of the turbojet engine become available for cruising conditions.

Supersonic flight speeds. - A plot of airplane load-carrying capacity and rate of fuel consumption per gross airplane weight for supersonic flight conditions is presented in figure IV-9. The rapid increase in the net thrust of this engine with flight speed results in an increase in load-carrying capacity with an increase in flight speed for all altitudes. The fuel consumption per gross airplane weight decreases considerably at all flight speeds as the altitude is increased. These characteristics cause the maximum range to occur at the highest speed and altitude considered. This maximum initial range, which occurs at 1800 miles per hour and 50,000 feet altitude, is about 1900 miles.

The combined frontal area of the engines assumed for the computations of this plot was the same as for the turbojet engine, that is, 12.5 square feet. The gross weight of the airplane for this engine size for each altitude and flight speed considered is given in the following table:

Altitude (ft)	0	30,000	50,000
Flight speed (mph)	Gross weight (lb)		
1000	47,700	44,600	26,500
1500	83,500	93,300	57,100
1800	116,300	133,400	84,500

In order to illustrate the effect of airplane gross weight on the
flight range, the computations were repeated for different engine
sizes providing gross weights of 50,000 and 200,000 pounds at a
speed of 1800 miles per hour at 50,000 feet; the results are
included in figure IV-9. The disposable load per airplane gross
weight is nearly independent of the gross weight of the airplane
and the range is reduced about 20 percent as the gross airplane
weight is reduced from 200,000 to 50,000 pounds.

The load-range characteristics for the turbo-ram-jet engine
are compared with the turbojet engine in figure IV-10. For super-
sonic flight conditions, where the airplane drag is very high, the
high thrust of the turbo-ram-jet engine provides a greater load-
carrying capacity than the turbojet engine and has about the same
fuel consumption. This greater load-carrying capacity of the
turbo-ram-jet engine is primarily a result of the greater thrust
per engine weight than provided by the turbojet engine because
the exhaust-gas temperatures are not limited to a maximum turbine-
inlet temperature of about 1500° F. Based on the assumptions of
this analysis, the maximum range of an airplane powered by a
turbo-ram-jet engine at 1800 miles per hour is about 60 percent
greater than that provided by the turbojet engine at a speed of
1400 miles per hour. The turbo-ram-jet engine therefore provides
both a greater range and a greater load-carrying capacity than the
turbojet engine for supersonic speeds up to 1800 miles per hour
where it becomes equivalent in operation to a ram-jet engine.
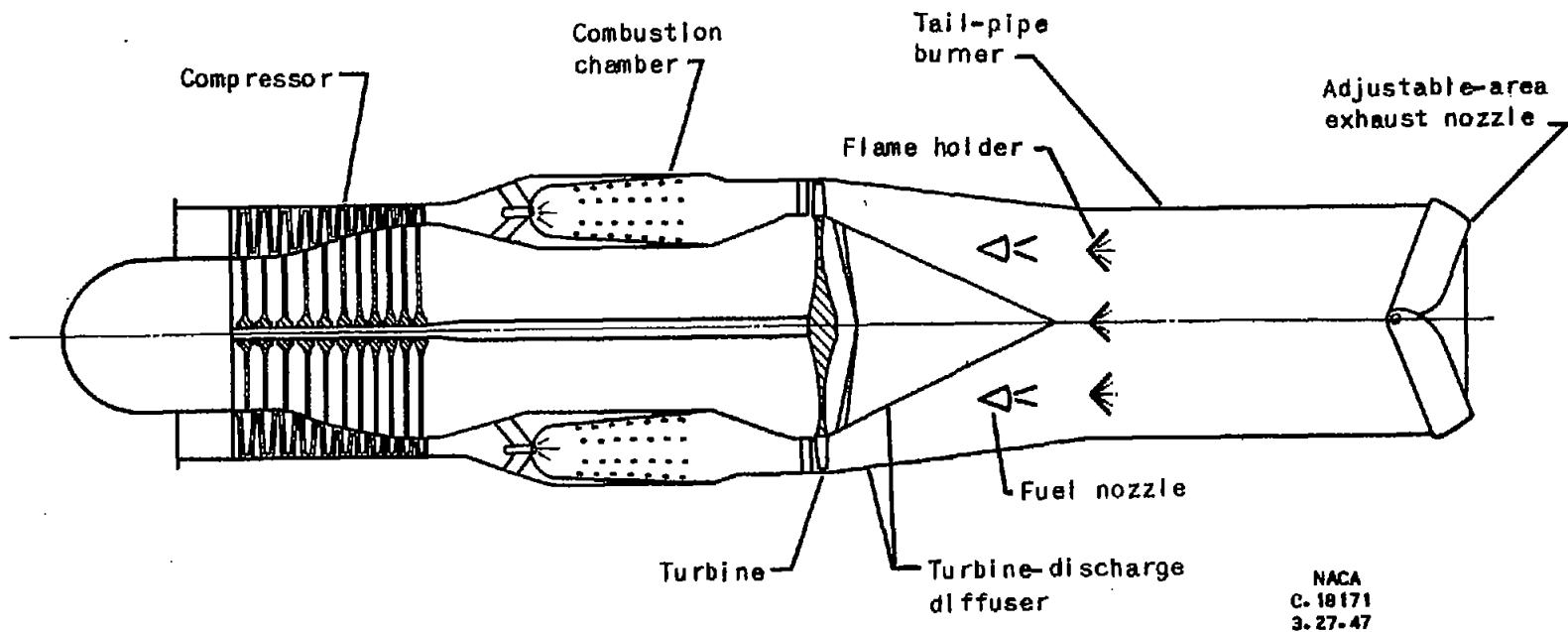
Compressor

Combustion
chamber

Tail-pipe
burner

Adjustable-area
exhaust nozzle

Flame holder

Fuel nozzle

Turbine

Turbine-discharge
diffuser

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Figure IV-1. - Diagrammatic sketch of turbo-ram-jet engine.

Fig. IV-1

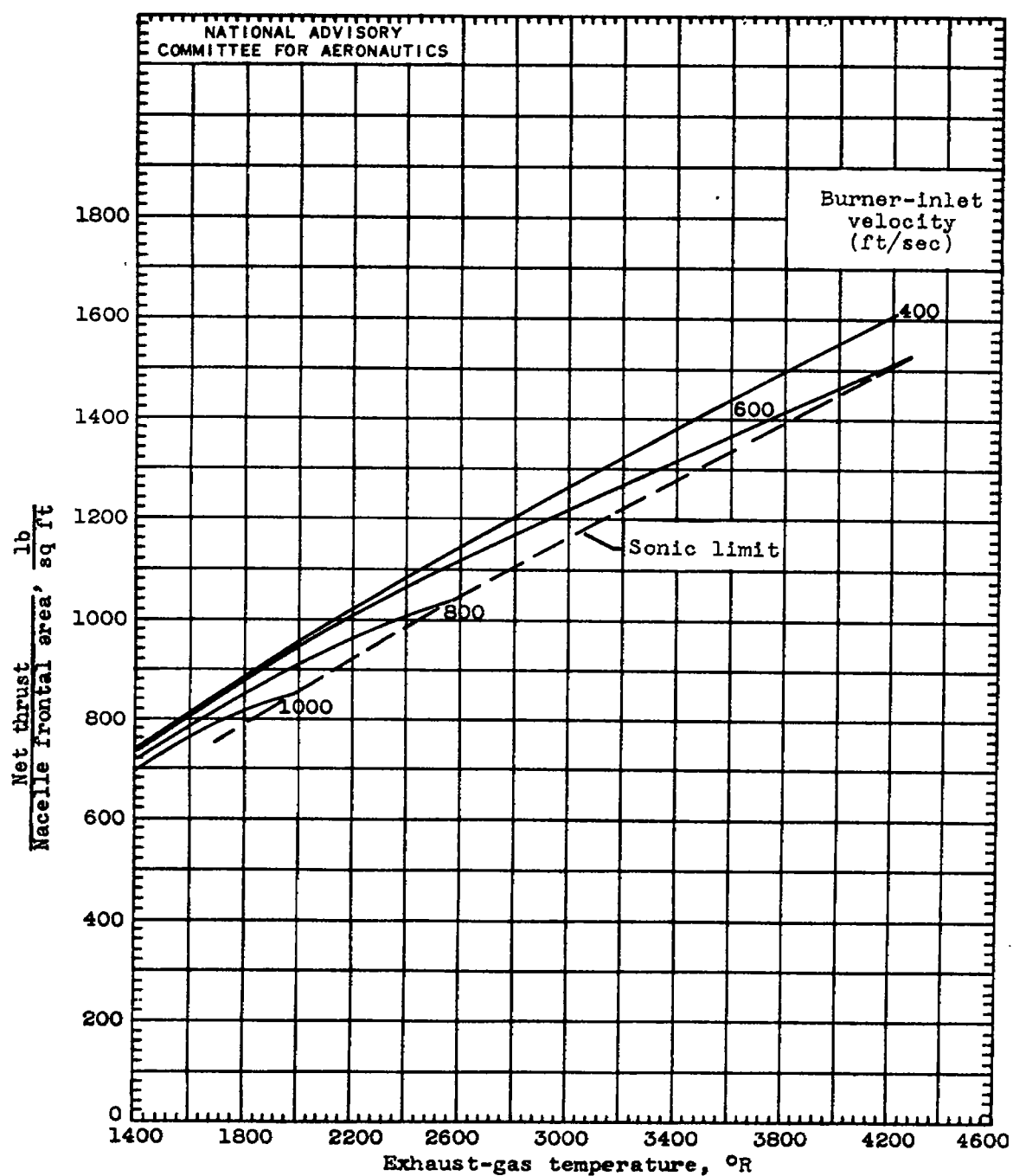Fig. IV-2   NACA TN No. 1349

Figure IV-2. - Variation of net thrust per unit nacelle frontal area with burner-inlet gas velocity and exhaust-gas temperature for turbo-ram-jet engine. Flight speed, 500 miles per hour; altitude, sea level.
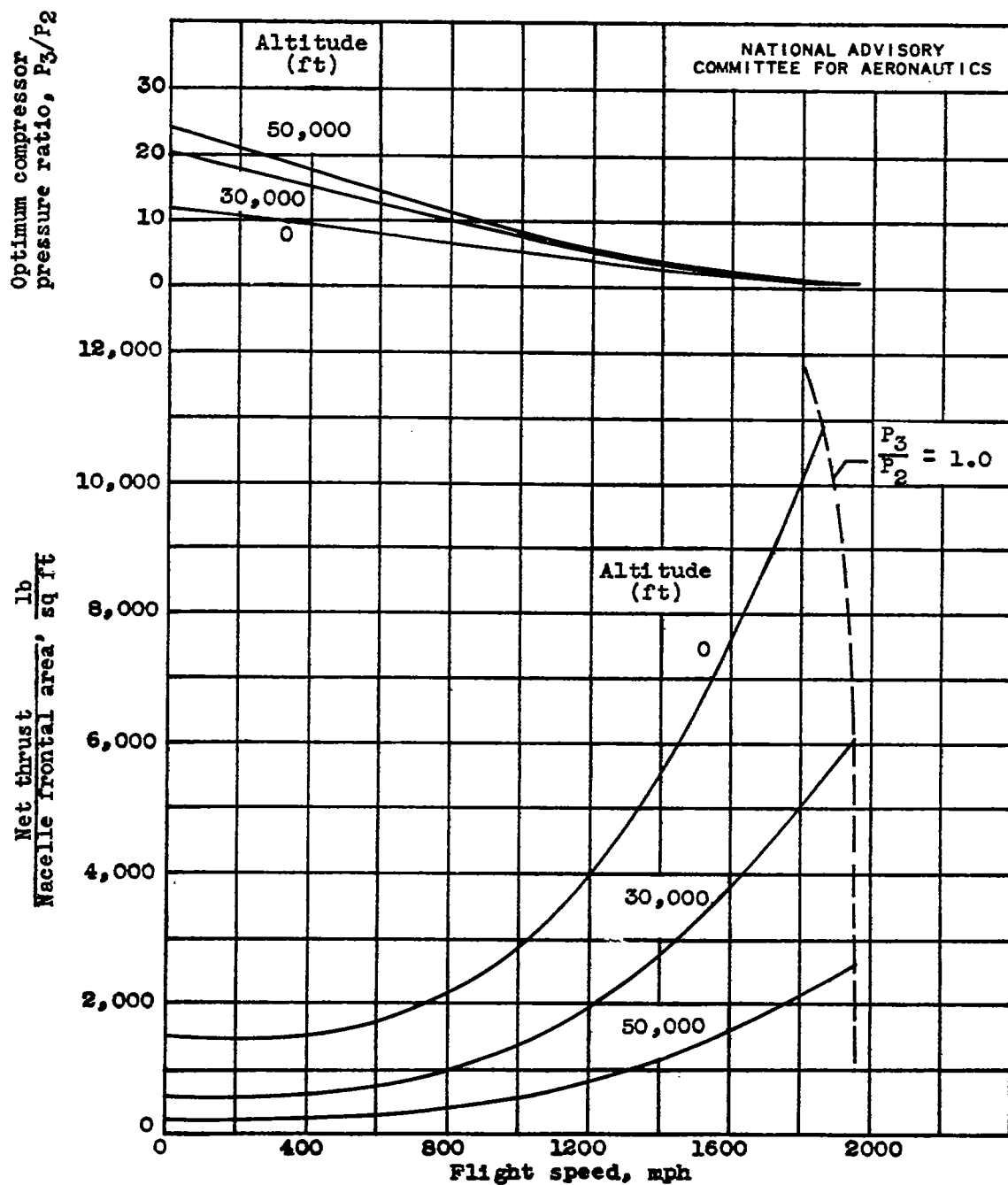
Figure IV-3. - Variation of net thrust per unit nacelle frontal area and optimum compressor pressure ratio with flight speed and altitude for turbo-ram-jet engine. Stoichiometric fuel-air ratio.

Fig. IV-4   NACA TN No. 1349

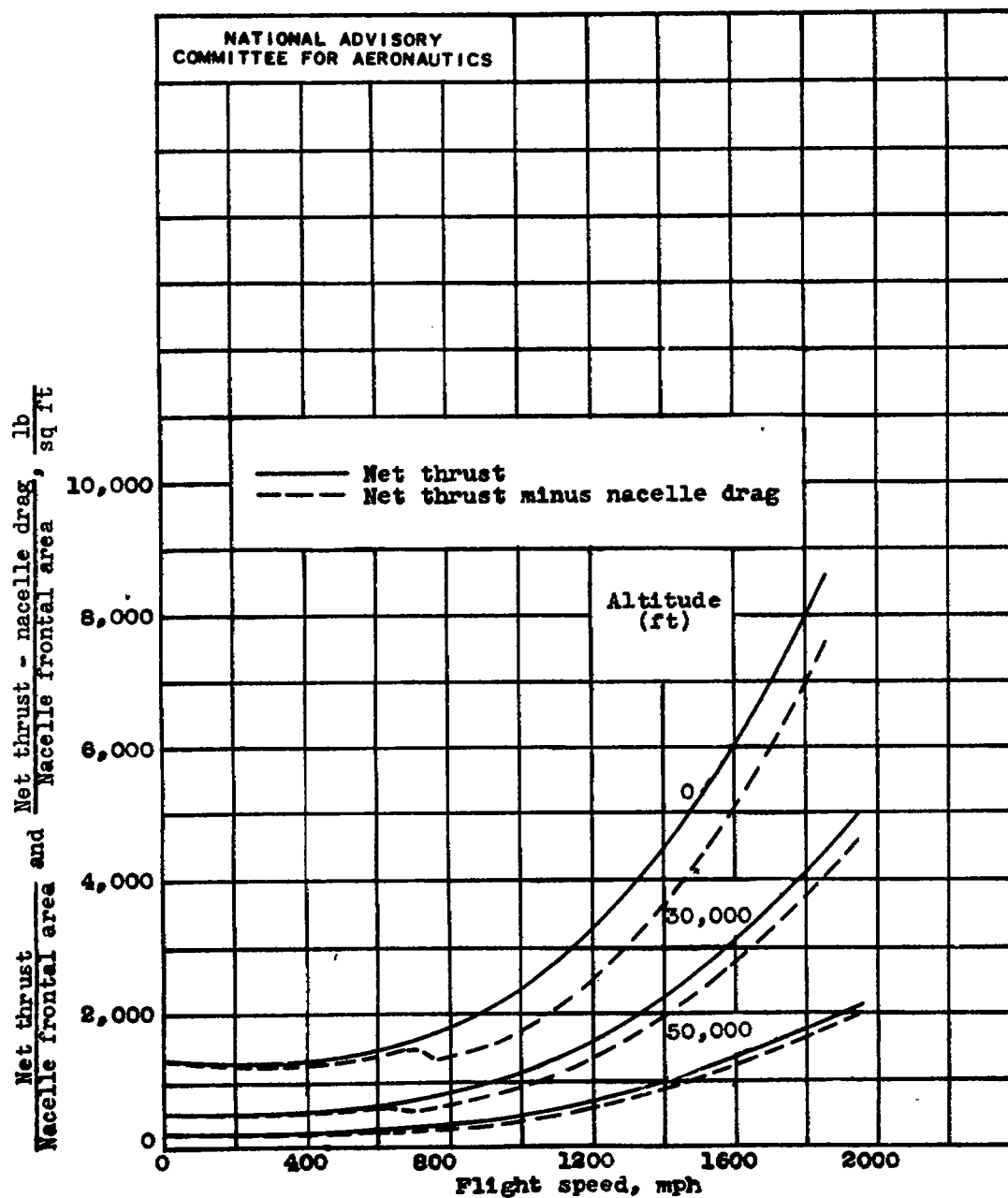Figure IV-4. - Variation of net thrust and net thrust minus nacelle drag per unit nacelle frontal area with flight speed and altitude for turbo-ram-jet engine. Over-all fuel-air ratio, 0.045.
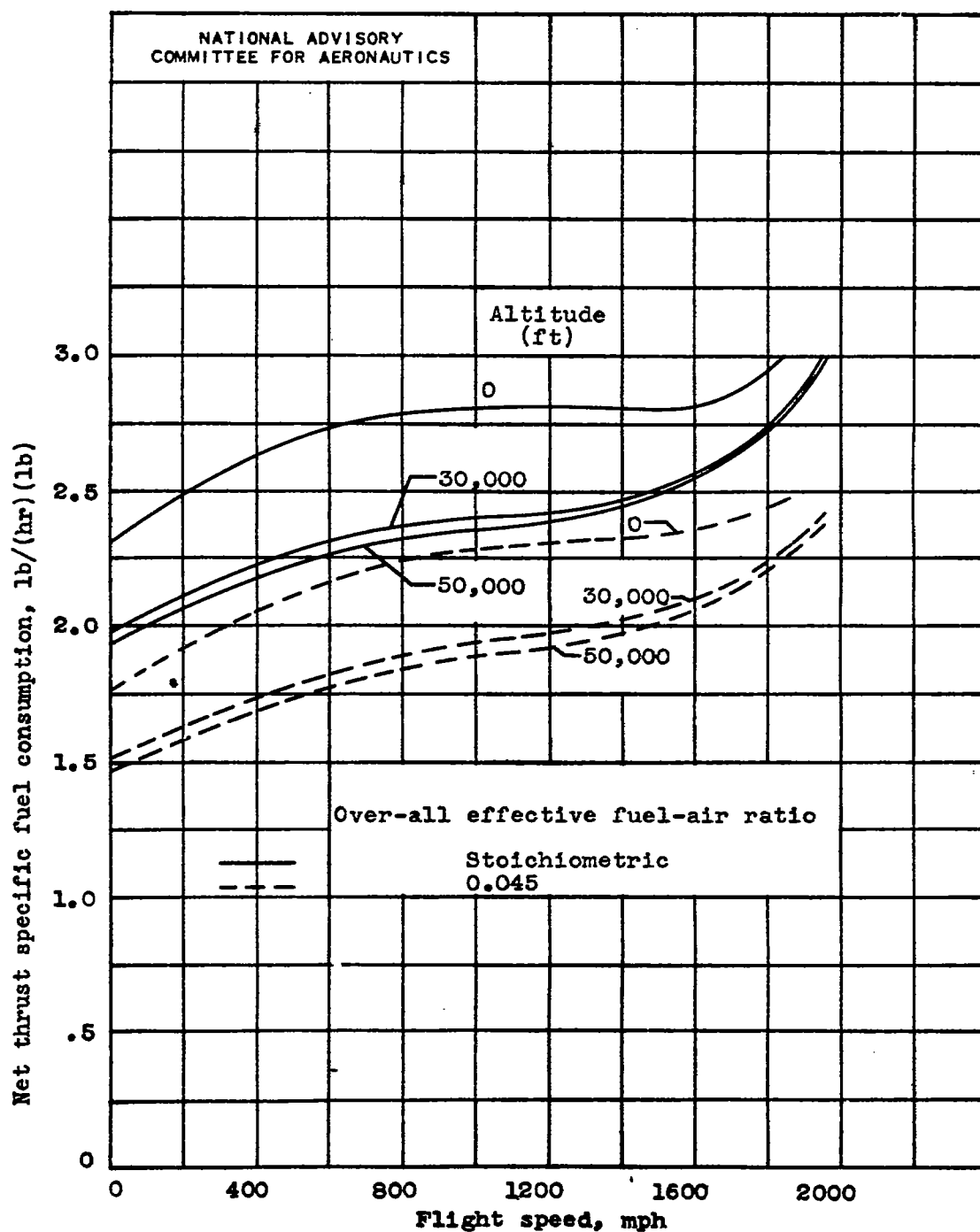
Figure IV-5. - Variation of net thrust specific fuel consumption with flight speed and altitude for turbo-ram-jet engine. Primary combustion efficiency, 95 percent; secondary combustion efficiency, 90 percent.

Fig. IV-6   NACA TN No. 1349
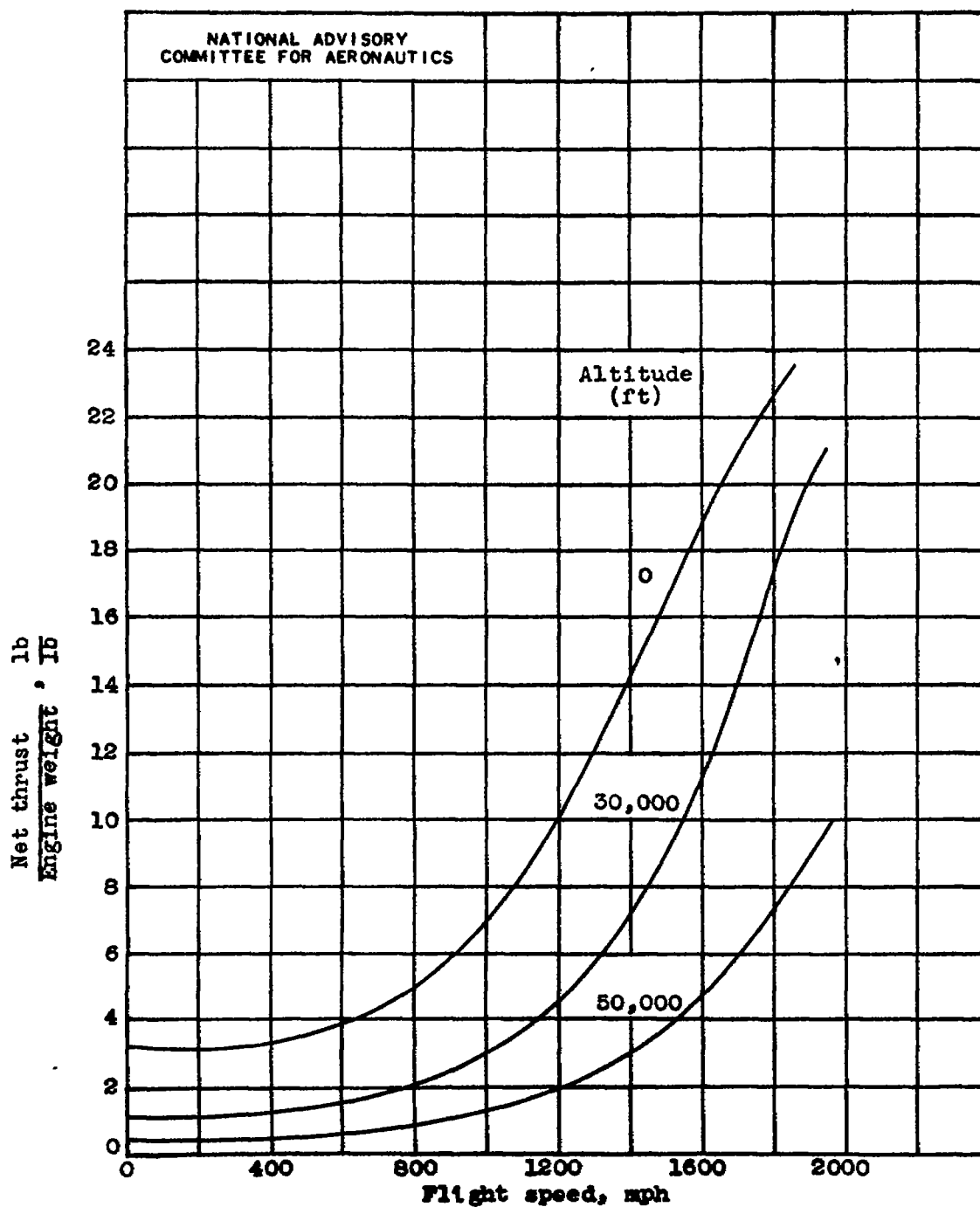
Figure IV-6. - Variation of net thrust per unit engine weight with flight speed and altitude for turbo-ram-jet engine. Over-all fuel-air ratio, 0.045.

Fig. IV-7a
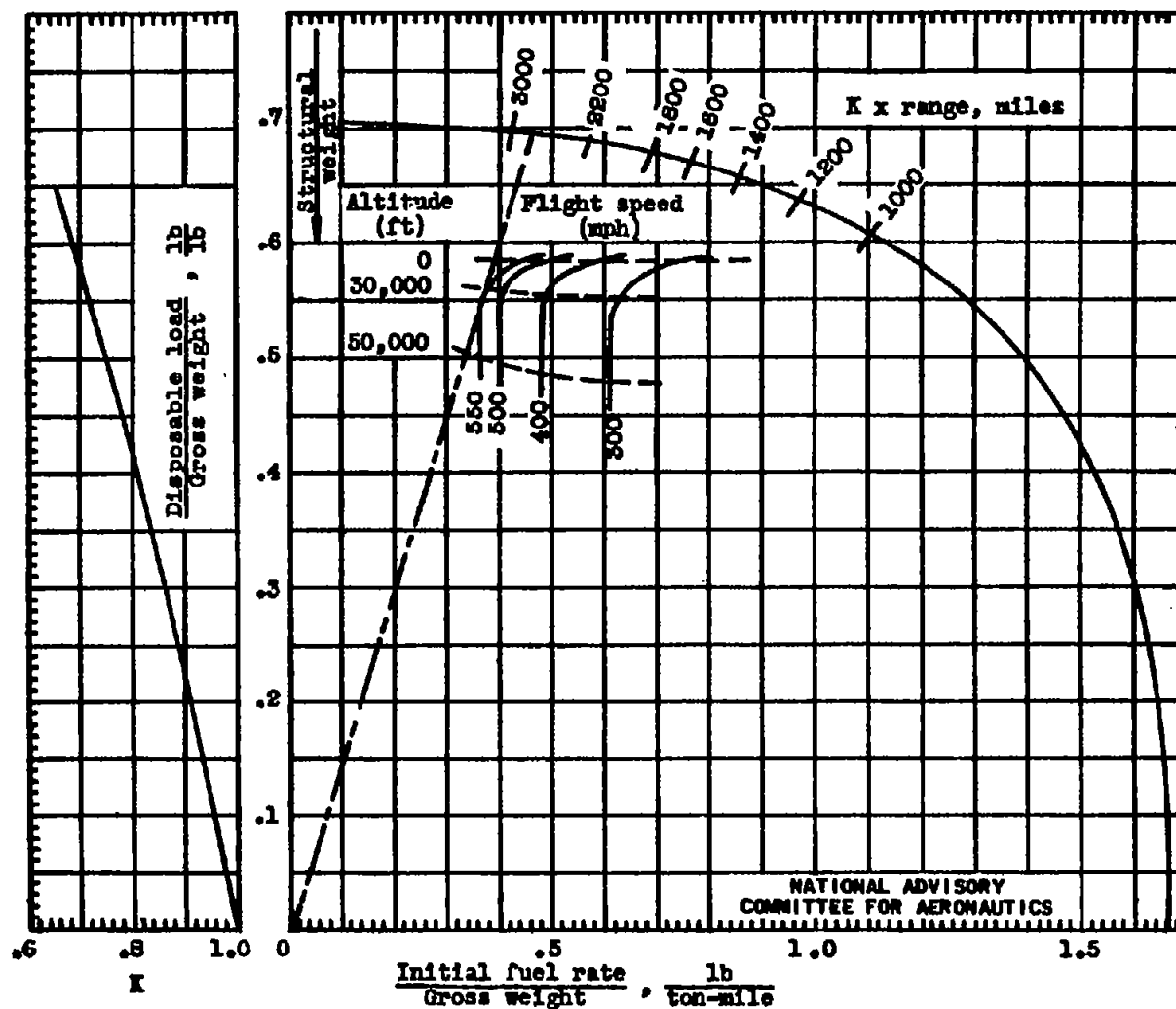
(a) Constant L/D, 18.

Figure IV-7. - Load-range characteristics of turbo-ram-jet engine at subsonic flight
speeds. Over-all fuel-air ratio, 0.045.

Fig. IV-7b
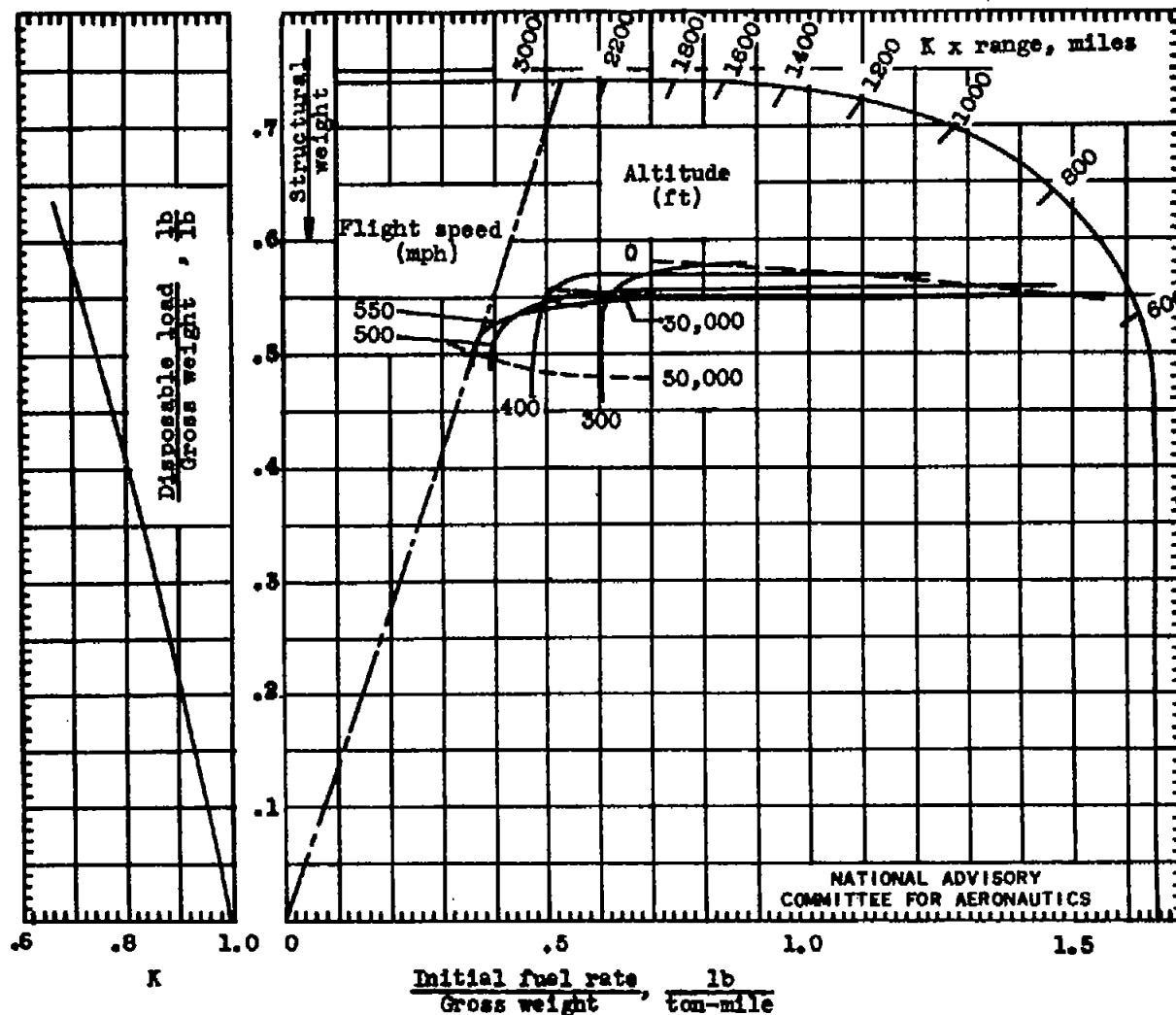
(b) Wing loading limited to 80 pounds per square foot

Figure IV-7. – Concluded. Load-range characteristics of turbo-ram-jet engine at subsonic flight speeds. Over-all fuel-air ratio, 0.045.

Figure IV-8. - Comparison of load-range characteristics of compound, turbine-propeller, turbojet, and turbo-ram-jet engines at subsonic flight speeds. Wing loading limited to 80 pounds per square foot.

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Fig. IV-8

Fig. IV-9

Figure IV-9. – Load-range characteristics of turbo-ram-jet engine at supersonic flight speeds. Over-all fuel-air ratio, 0.045.

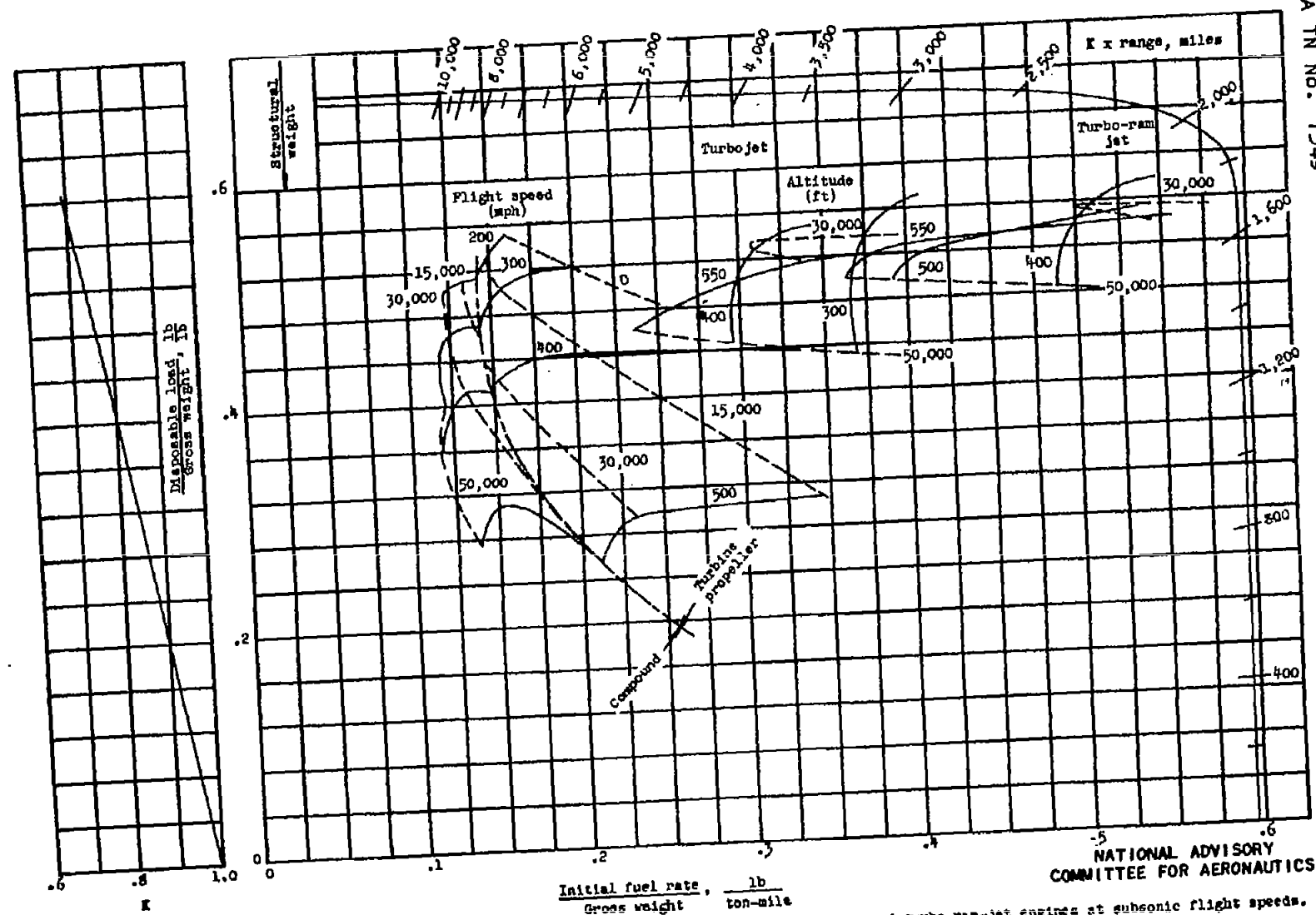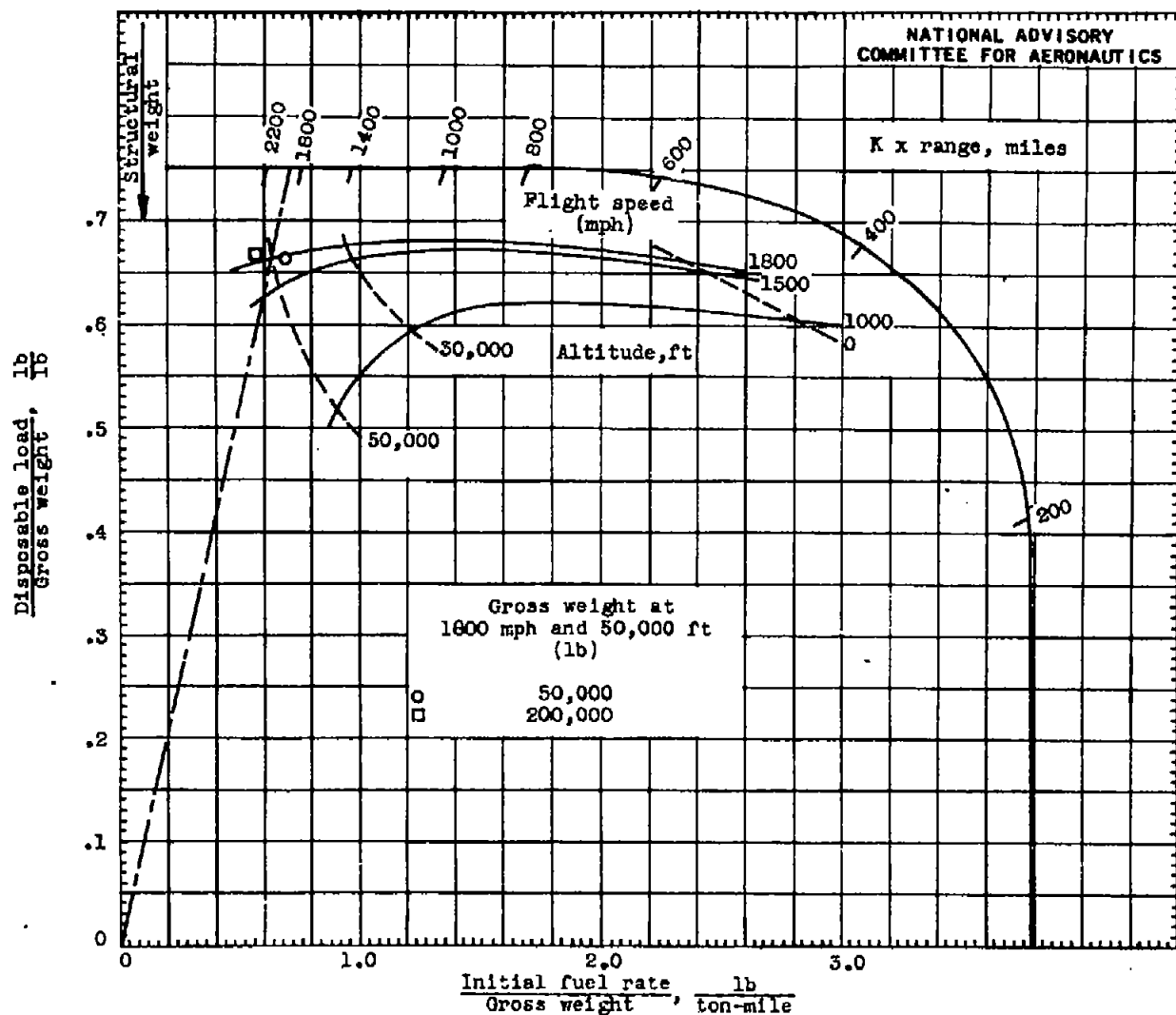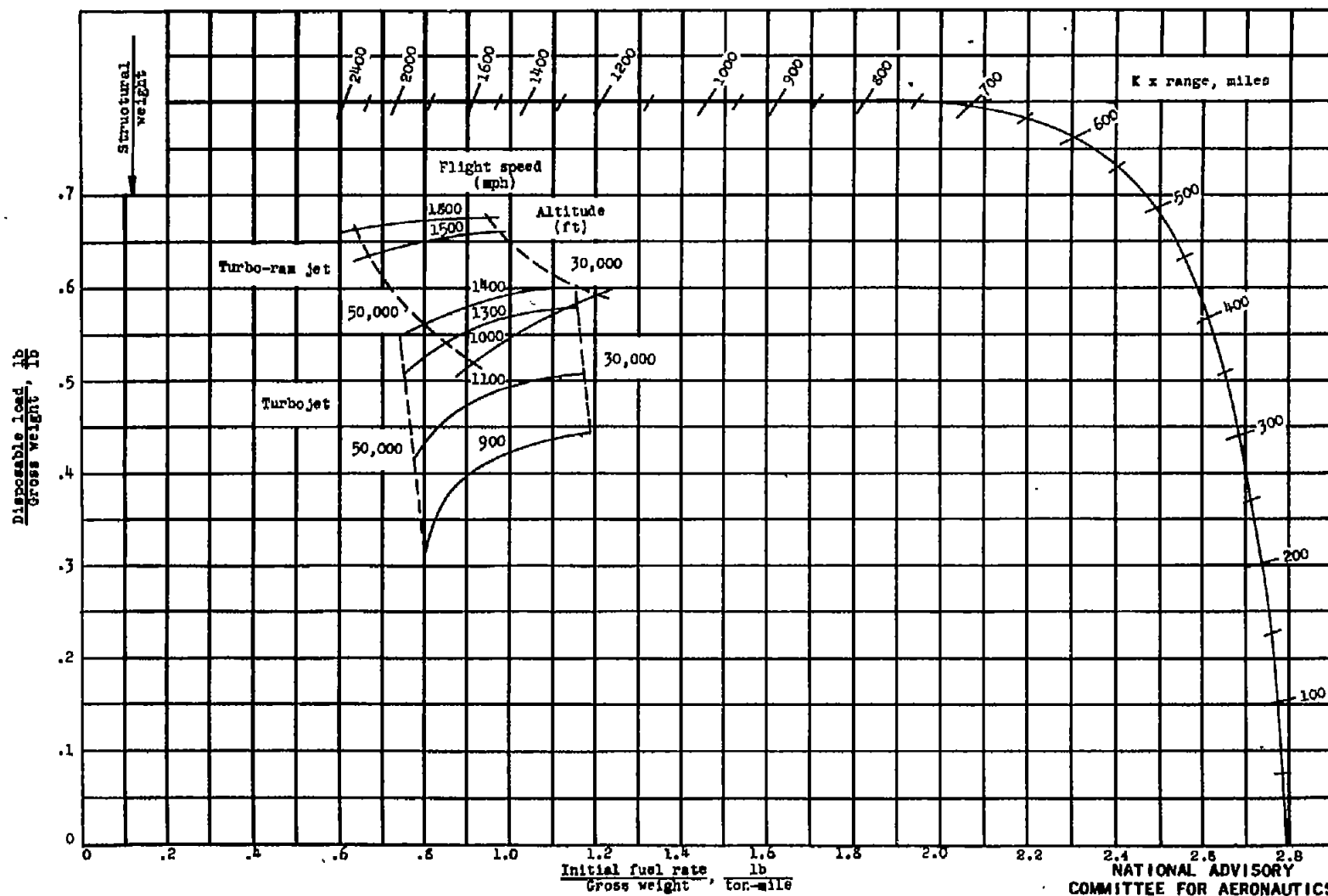Figure IV-10. - Comparison of load-range characteristics of turbojet and turbo-ram-jet engines at supersonic flight speeds.

Fig. IV-10

V - THE RAM-JET ENGINE

Description

The ram-jet engine (fig. V-1) consists of a diffuser in which
the air is compressed from free-stream static pressure to a pres-
sure somewhat lower than free-stream total pressure, a combustion
chamber in which fuel is burned, and an exit nozzle through which
the gases expand.

Engine Performance

For this analysis, the three most important criterions for
evaluating the performance of aircraft engines are: efficiency or
fuel economy, thrust per unit engine frontal area, and thrust per
unit engine weight. The efficiency of the ram-jet engine, like
that of all heat engines, increases with compression ratio. At
subsonic flight speeds, the ram compression is so low that the ram
jet cannot compete with other engines except perhaps where engine
cost and simplicity are of great importance. At supersonic flight
speeds, however, the ram compression is considerable and high
efficiencies are obtainable. Because of the simplicity of the
engine, the ram jet develops greater thrust per unit weight than
the engines previously discussed except at low flight speeds. The
thrust per unit frontal area increases both with efficiency and
air flow through the engine; therefore, much greater values of
thrust per unit area are obtainable at the higher airspeeds. The
best performance of the ram-jet engine is therefore obtained at
high flight speeds.

The variation of net thrust per unit engine frontal area and
net thrust specific fuel consumption with fuel-air ratio and
combustion-chamber inlet velocity for a ram jet burning gasoline
is shown in figure V-2 for a flight speed of 1150 miles per hour
at sea level (Mach number, 1.5). The data shown are for a combus-
tion efficiency of 100 percent. Underexpanding exit nozzles have
been used in the calculations where use of a completely expanding
exit nozzle would have resulted in a larger exit area than
combustion-chamber area. The performance at combustion efficiencies
other than 100 percent may be obtained by dividing the fuel-air
ratio and specific fuel consumption shown in figure V-2 by the
actual combustion efficiency in order to determine the actual fuel-
air ratio and specific fuel consumption. In general, the thrust
per unit engine frontal area increases with increasing fuel-air

ratio and combustion-chamber velocity up to the choking line. The
best fuel economy for any particular thrust per unit engine frontal
area is obtained at a combustion-chamber inlet velocity slightly
lower than would be required for choking in the combustion chamber.
The specific fuel consumption, of course, decreases with decreasing
thrust per unit frontal area. Inasmuch as the exit area is equal
to the combustion-chamber area near the choking line and the dif-
ference between the inlet area and combustion-chamber area is small
compared with that at lower combustion-chamber velocities, the
external pressure drag will also be comparatively low. The region
just to the left of the choking line is therefore the region of
optimum performance for the ram jet at a flight Mach number of 1.5.
The maximum thrust per unit engine frontal area obtainable is
approximately 3000 pounds per square foot, which is about 20 percent
lower than that obtainable with the turbo-ram-jet engine at the
same speed and altitude. The fuel consumption at maximum thrust is
about 3.3 pounds per hour per pound of thrust, which is about
23 percent above the fuel consumption of the turbo-ram-jet at a
flight speed of 1150 miles per hour.

The variation in net thrust per unit engine frontal area and
fuel consumption with fuel-air ratio and combustion-chamber velocity
for a higher flight speed (2300 mph at sea level; Mach number, 3.0)
is shown in figure V-3. The highest thrust is obtained at a
combustion-chamber inlet velocity of 325 feet per second, which
corresponds to the point at which the inlet area is equal to the
combustion-chamber area. Increasing the combustion-chamber inlet
velocity beyond this point results in an inlet area larger than
combustion-chamber area, with a consequent decrease in thrust per
unit engine frontal area. For any particular value of thrust per
unit frontal area, the lowest fuel consumption is also obtained at
a combustion-chamber inlet velocity of 325 feet per second. The
external pressure drag is zero for this case because the inlet
area, combustion-chamber area, and outlet area are all equal. At
this flight speed, the optimum operating region is well away from
the choking line. The maximum thrust per unit engine frontal area
is extremely high, approximately 22,000 pounds per square foot.
At thrust values somewhat lower than maximum, fuel consumptions of
about 2 pounds per hour per pound of thrust corresponding to approx-
imately 0.33 pound of fuel per thrust horsepower-hour are obtainable,
which indicates that the engine is operating very efficiently at
this flight speed.

The variation in maximum net thrust per unit engine area with
flight speed and altitude is shown in figure V-4. These data were
calculated for an actual fuel-air ratio of 0.067, a combustion

efficiency of 90 percent, optimum combustion-chamber inlet velocity, and underexpanding exit nozzles. The thrust increases rapidly with flight speed and decreases with increasing altitude. As pointed out in the discussion of figure V-3, extremely high thrusts per unit frontal area are obtainable at high speeds at sea level.

Figure V-5 shows the thrust specific fuel consumption corresponding to the thrusts given in figure V-4. The fuel consumption decreases with increasing flight speed up to a speed of 2400 miles per hour. The fuel consumption also decreases with increasing altitude up to the tropopause (approximately 35,000 ft), above which it remains essentially constant. At very high altitudes (above 100,000 ft), the fuel consumption will again vary with altitude because of the variation in ambient-air temperature with altitude at these heights.

Load-Range Characteristics

The assumptions used in the analysis to evaluate the effect of altitude and flight speed upon the range of ram-jet-powered aircraft are outlined in the appendix. The type of aircraft considered has a ram-jet engine located at each wing tip and the fuel is stored in the fuselage. It was assumed that the combined frontal area of the two engines was 12.5 square feet. A combustion efficiency of 90 percent and the optimum combustion-chamber inlet velocity were also assumed. Performance curves similar to those shown in figures V-2 and V-3 were used to obtain operating points giving the longest range for the specified flight speed and altitude. In general, it was found that for the assumption used, the best ranges were obtained at fuel-air ratios from 0.03 to 0.05 and combustion-chamber inlet velocities from 180 to 400 feet per second.

The thrust per unit engine frontal area, the thrust minus engine drag per unit engine frontal area, the net thrust specific fuel consumption, and the thrust per unit engine weight used in computing the ranges are shown in figure V-6. In estimating the engine weights, it was assumed that the engine length-diameter ratio was 8 and skin thicknesses necessary to provide reasonable hoop stresses were calculated. At low flight speeds and high altitudes where the required skin thicknesses gave an engine weight lower than 700 pounds for a frontal area of 12.5 square feet, an engine weight of 700 pounds was used.

The ratio of disposable load to gross weight and the initial fuel rate per mile per ton gross weight is shown in figure V-7 for

a range of flight speeds from 1000 to 2500 miles per hour and alti-
tudes from sea level to 100,000 feet. A K × range scale and
broken guide lines indicating the location of the origin are marked
for convenience in estimating the range. It may be seen that the
range increases with altitude and increases with flight speed up
to 2000 miles per hour. At an altitude of 50,000 feet, the range at
2500 miles per hour is somewhat lower than that at 2000 miles per
hour; whereas, at an altitude of 100,000 feet, the ranges at these
two flight speeds are approximately the same. At altitudes lower
than 50,000 feet, the range at 2500 miles per hour was found to be
appreciably lower than that at 2000 miles per hour. Although the
engine efficiency and thrust per unit engine frontal area increase
as the flight speed increases from 2000 to 2500 miles per hour, the
improvement in engine performance is too small to offset the
increased power required for flight at the higher flight speed.
At a flight speed of 2000 miles per hour, the range increases from
500 miles at sea level to 3800 miles at an altitude of 100,000 feet.
The large increase in range with increasing altitude occurs because
of the lower airplane drag at the higher altitudes due to the
lower air density.

The airplane gross weights calculated for the various flight
conditions shown in figure V-7 are given in the following table:

Altitude (ft) Flight speed (mph)	~ 0	30,000	50,000	100,000
	Gross weight (lb)			
1000	26,000	20,000	11,700	-------
1500	73,000	63,200	40,800	-------
2000	155,000	123,200	81,000	12,000
2500	-------	-------	75,600	14,700

The effect of varying the airplane size upon the range has been
investigated for a flight speed of 2000 miles per hour and an alti-
tude of 50,000 feet. The airplane gross weight for the case orig-
inally calculated for this flight condition (engine frontal area,
12.5 sq ft) was 81,000 pounds. It was found that by increasing the
gross weight from 81,000 to 200,000 pounds the range was increased
about 15 percent. A decrease in gross weight to 50,000 pounds
decreased the range about 5 percent.

The flight speed giving the longest range shown in figure V-7 (2000 mph) is replotted in figure V-8 to give a comparison with the turbojet and turbo-ram-jet engines. The range obtainable with the ram jet at 2000 miles per hour and 50,000 feet is somewhat greater than that obtainable at supersonic speeds at this altitude with either the turbojet or the turbo-ram-jet engine, and is closely approached by the turbo-ram-jet engine only at the higher flight speed (1800 mph) where the turbo-ram-jet engine is operating essentially as a ram jet because of the low compressor pressure ratio at this flight speed. If kerosene had been used as the fuel in the ram-jet calculations, as was done for the turbo-ram-jet engine, the range at 2000 miles per hour and 50,000 feet would have been approximately 5 percent greater than that shown in figures V-7 and V-8 due to the greater density of the kerosene.

Diffuser

Combustion
chamber

Exit
nozzle

Flame holder

Air
flow

Fuel nozzle
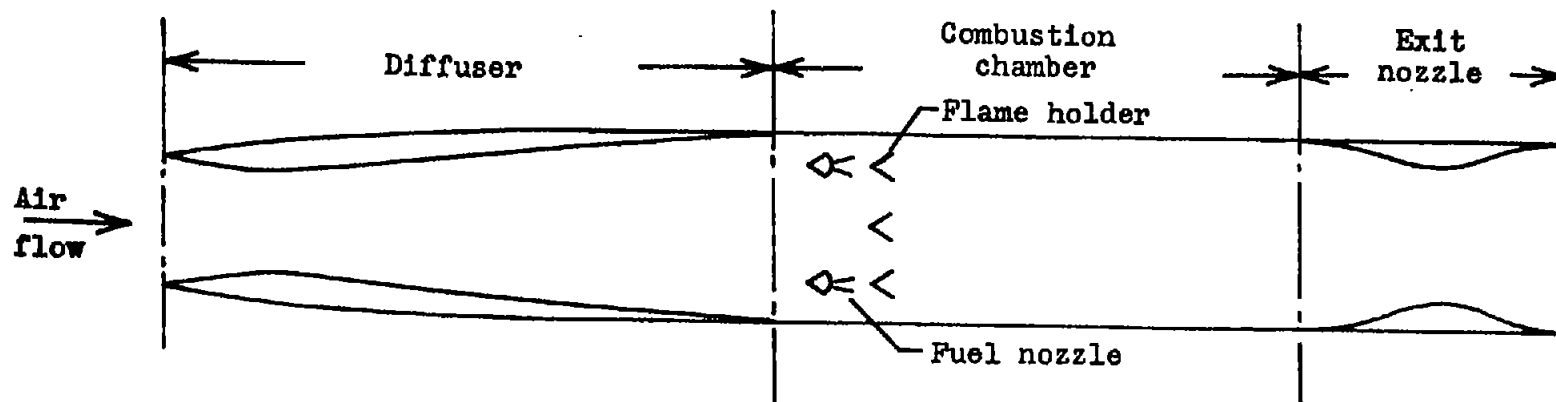
Figure V-1. - Diagrammatic sketch of ram-jet engine.

Fig. V-1

Fig. V-2   NACA TN No. 1349
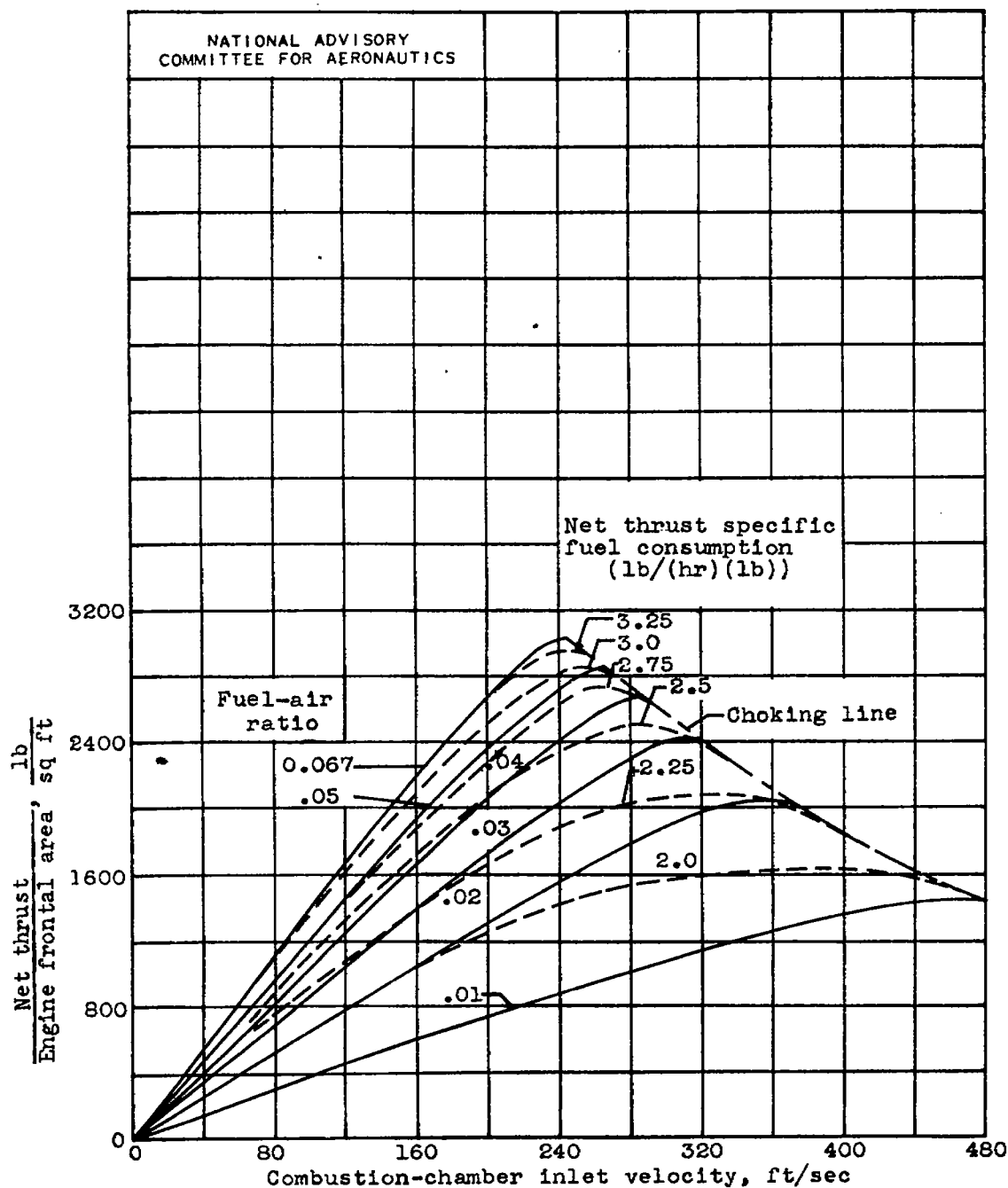
Figure V-2. - Variation of net thrust per unit engine frontal area with combustion-chamber inlet velocity for ram-jet engine at flight speed of 1150 miles per hour at sea level. Fuel, gasoline; combustion efficiency, 100 percent.

Fig. V-3
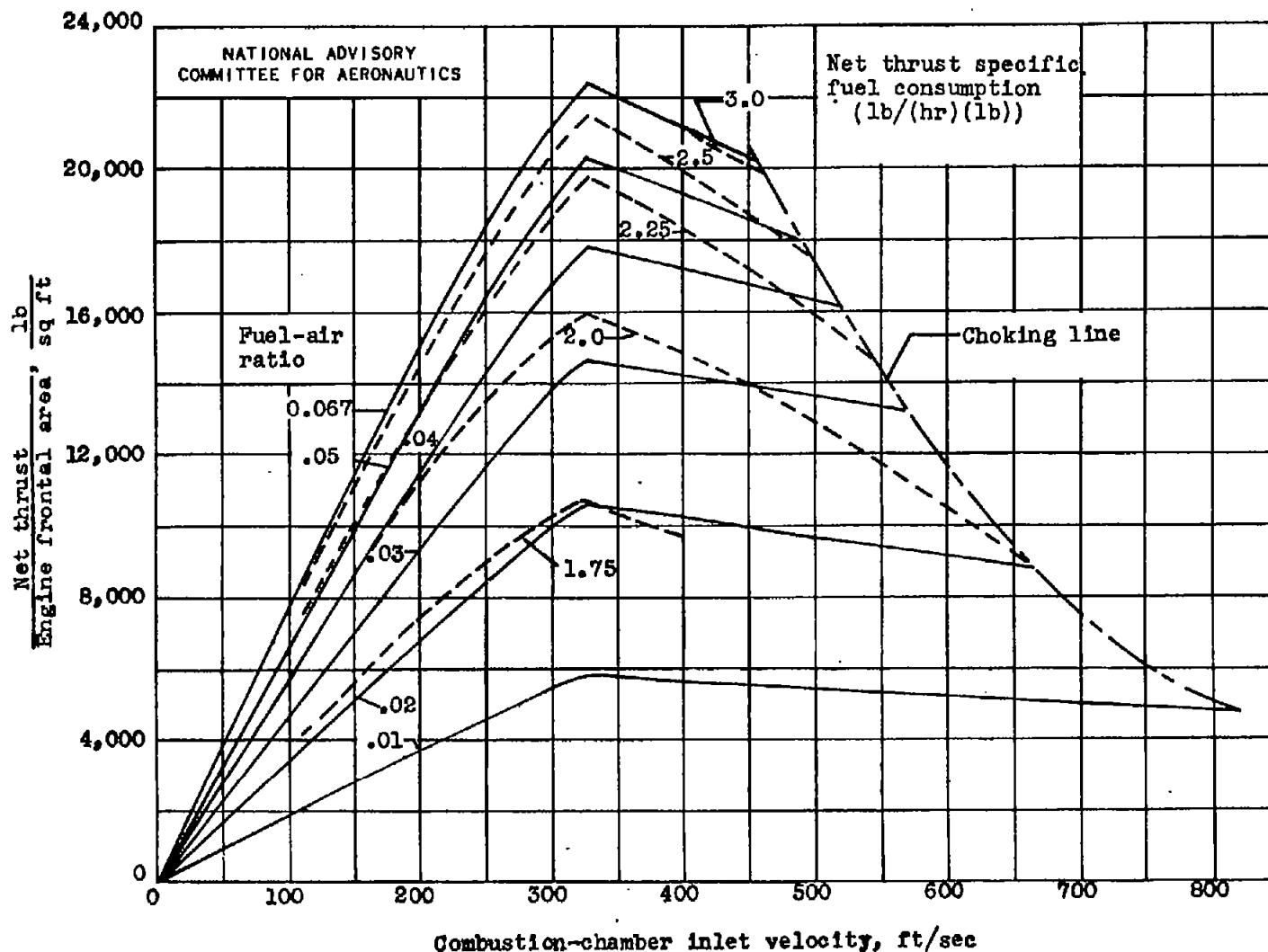
Figure V-3. - Variation of net thrust per unit engine frontal area with combustion-chamber inlet velocity for ram-jet engine at flight speed of 2300 miles per hour at sea level. Fuel, gasoline; combustion efficiency, 100 percent.

Fig. V-4   NACA TN No. 1349
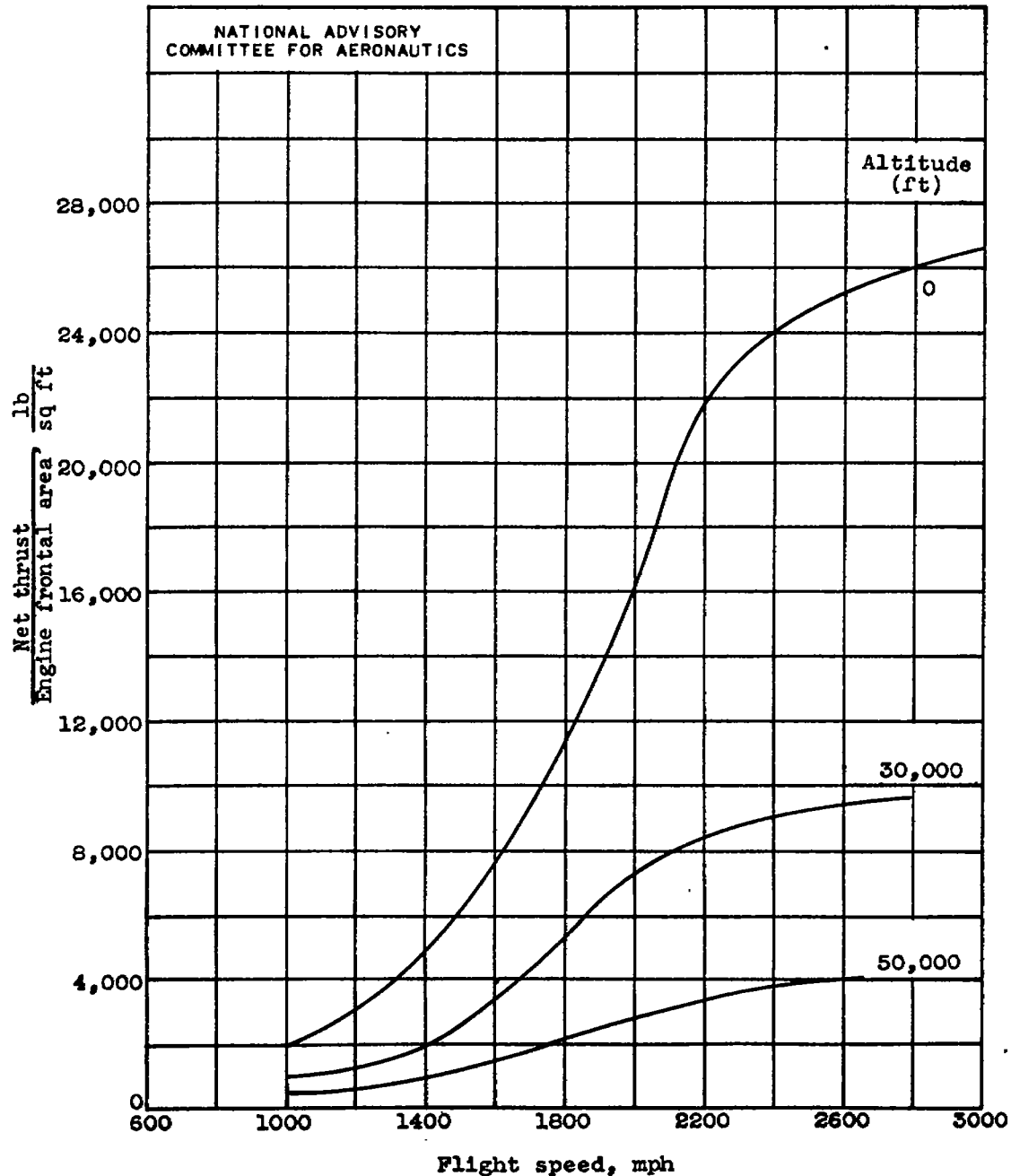
Figure V-4. - Variation of maximum attainable net thrust per unit engine frontal area with flight speed and altitude. Fuel, gasoline; combustion efficiency, 90 percent.
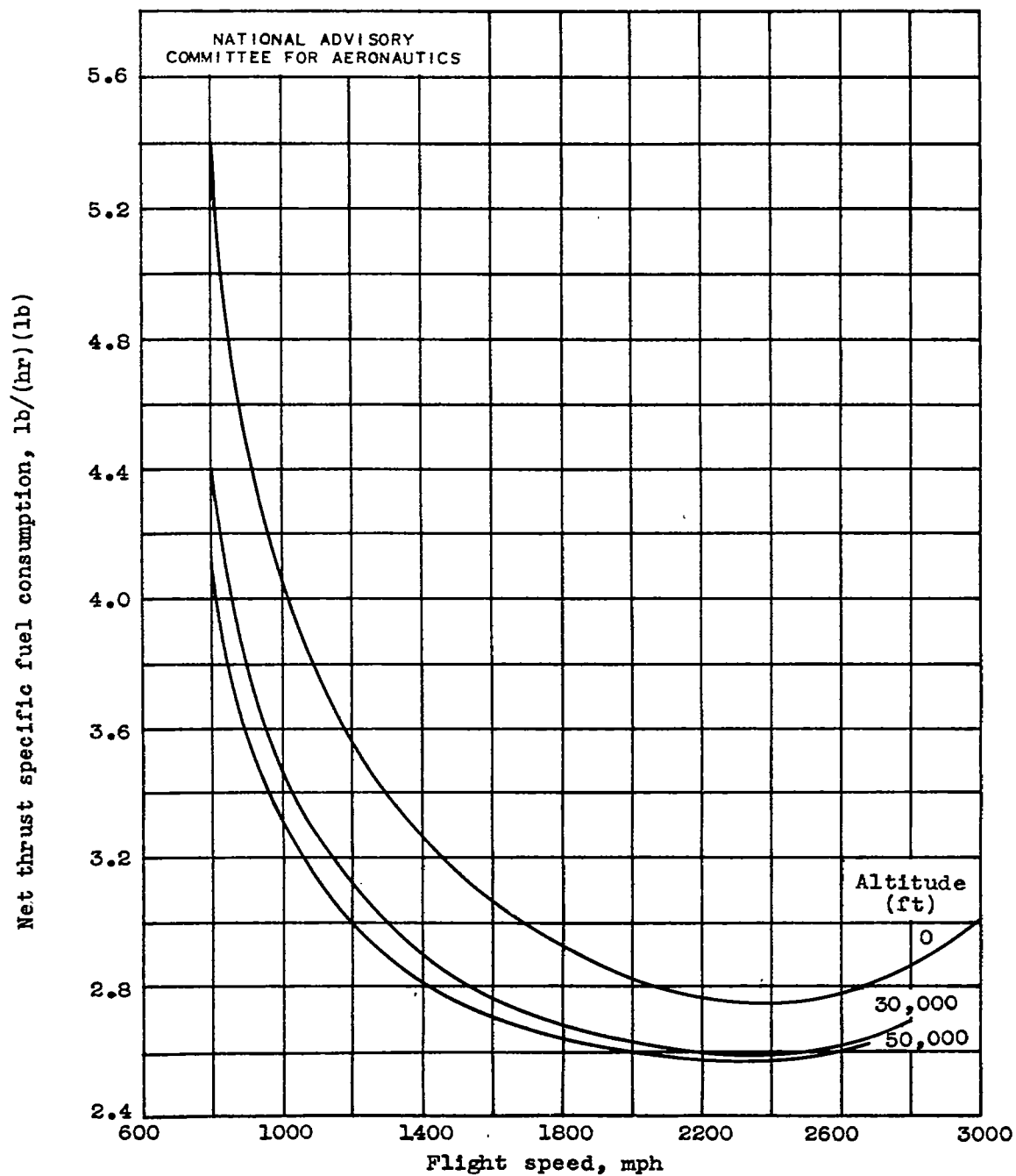
Figure V-5. - Variation of net thrust specific fuel consumption with flight speed and altitude for ram-jet engine developing maximum thrust. Fuel, gasoline; combustion efficiency, 90 percent.

Fig. V-6a   NACA TN No. 1349
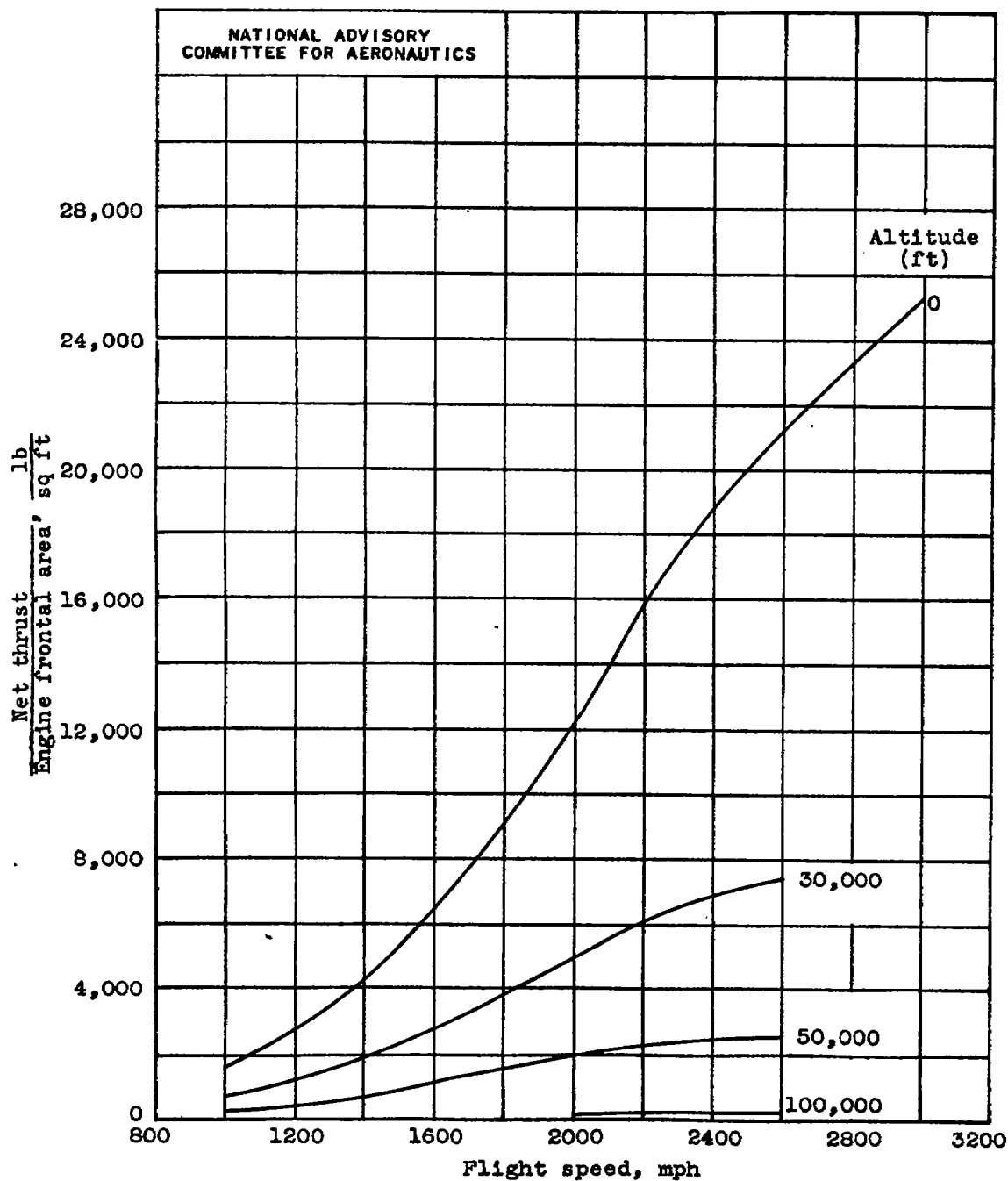
(a)   Variation of net thrust per unit engine frontal area with
flight speed and altitude.

Figure V-6. - Ram-jet performance used to determine maximum range.

Fig. V-6b
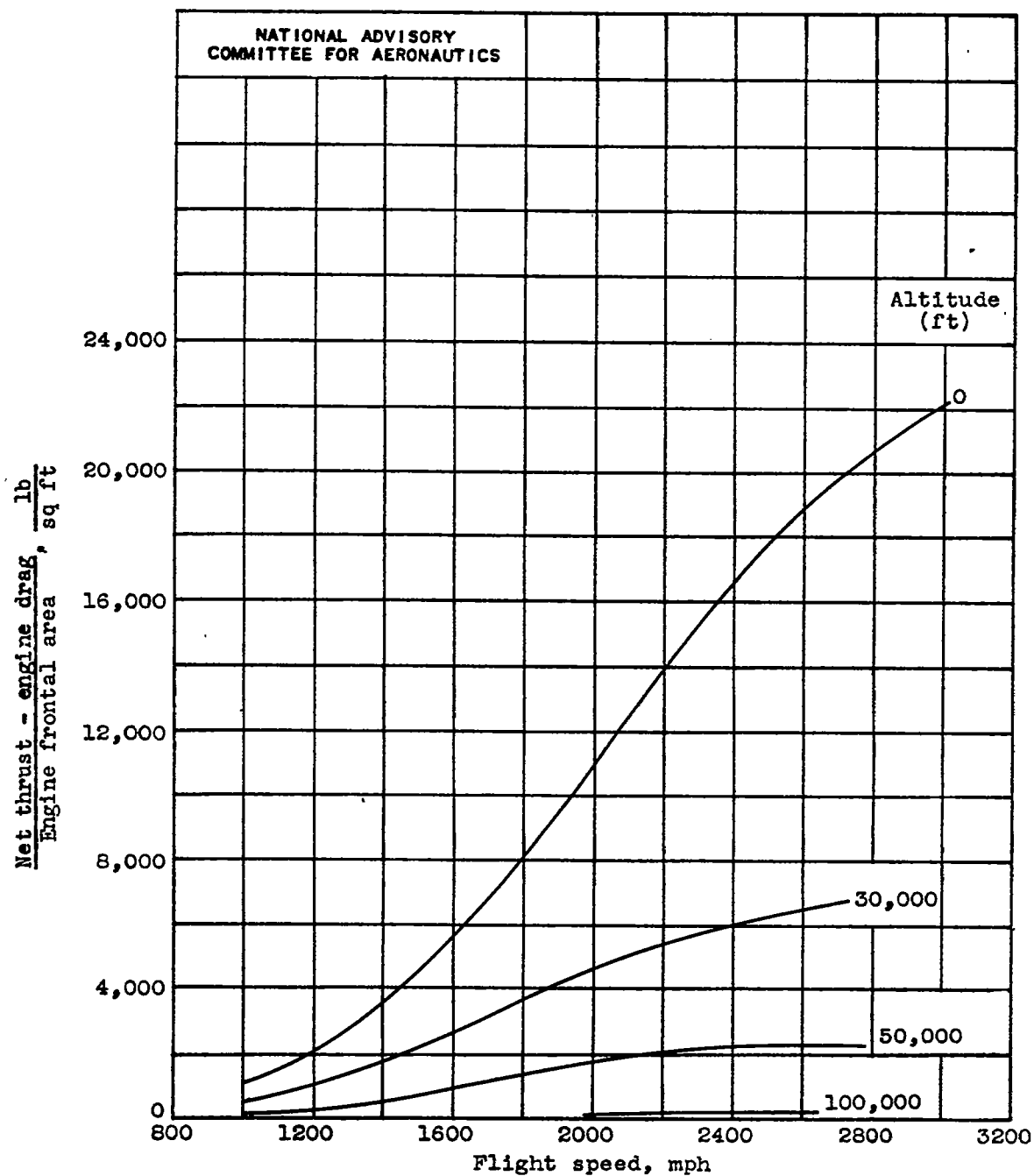
(b)   Variation of net thrust minus engine drag per unit engine
frontal area with flight speed and altitude.

Figure V-6. - Continued.   Ram-jet performance used to determine maximum
range.

Fig. V-6c   NACA TN No. 1349
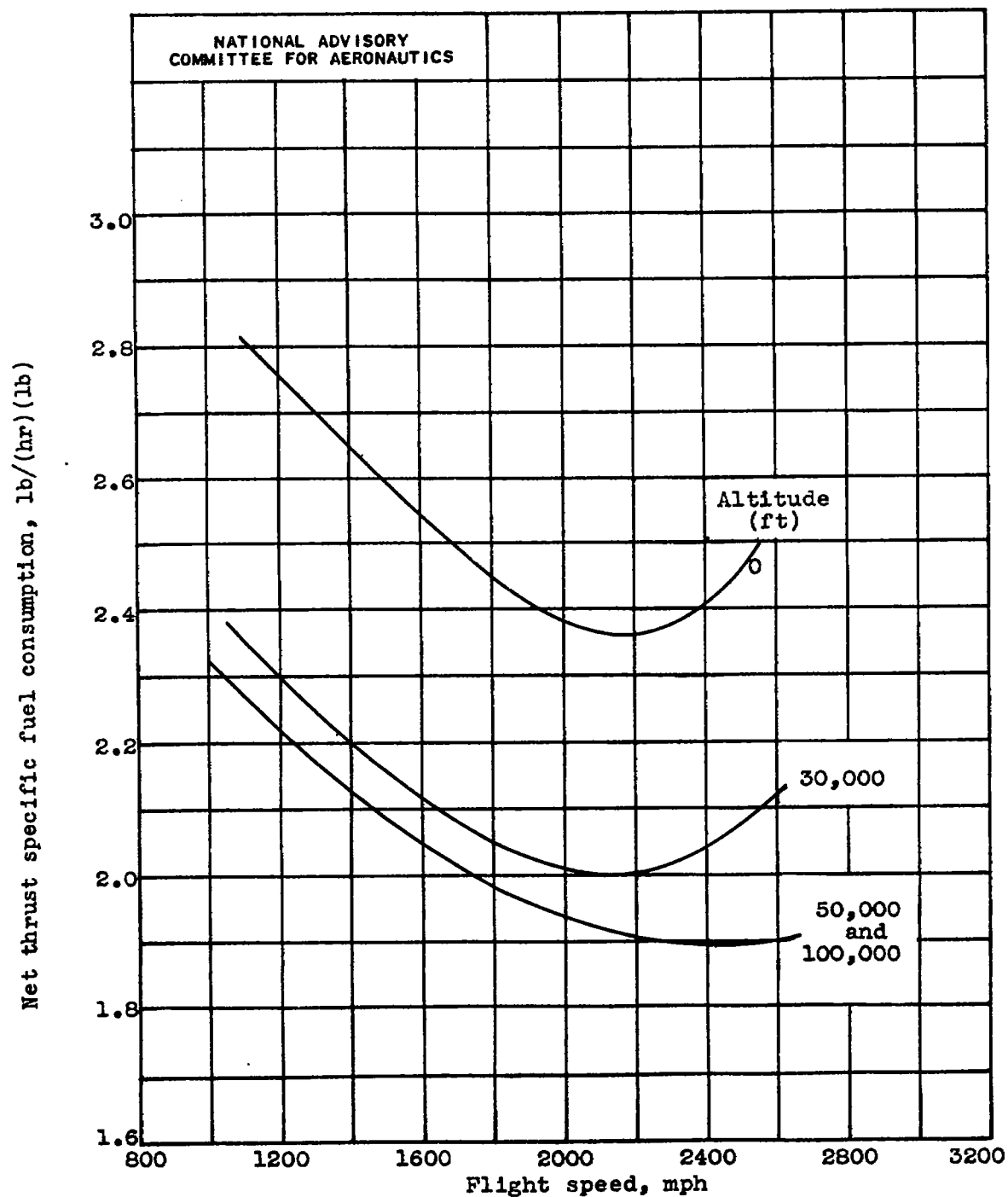
(c) Variation of net thrust specific fuel consumption with
flight speed and altitude.

Figure V-6. - Continued. Ram-jet performance used to determine maximum
range.
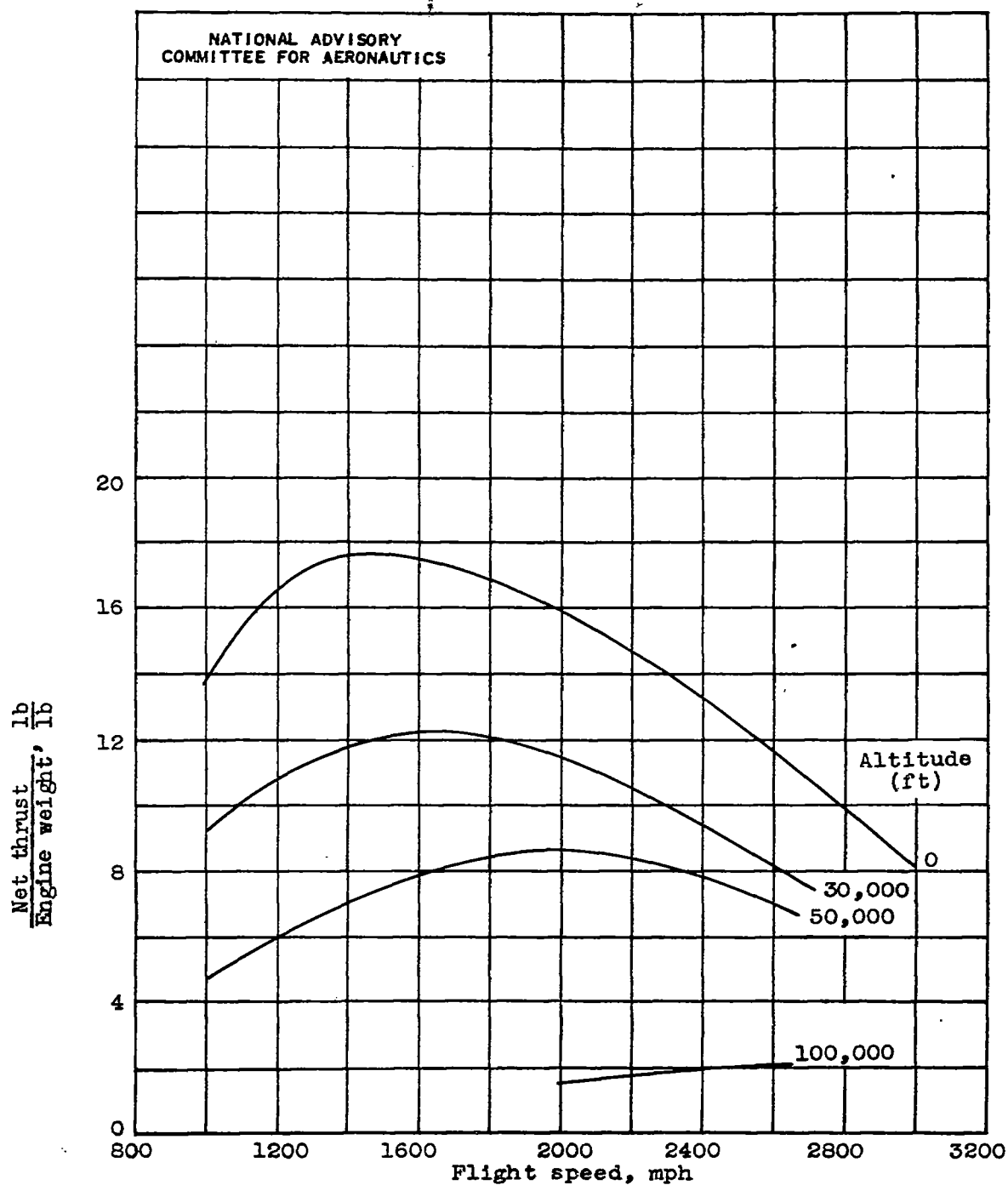
(d) Variation of net thrust per unit engine weight with flight speed and altitude.

Figure V-6. - Concluded. Ram-jet performance used to determine maximum range.
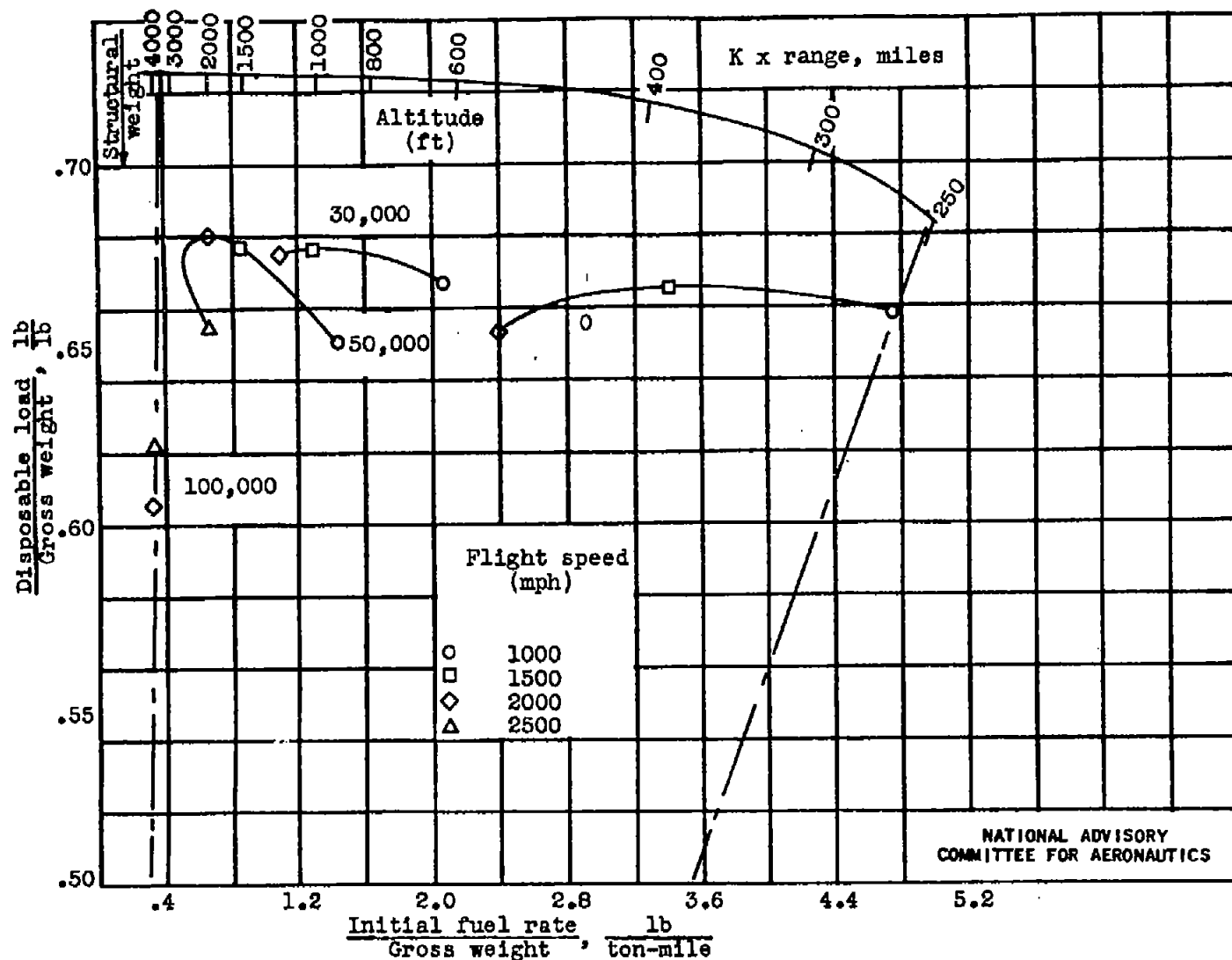
Fig. V-7

Figure V-7. - Load-range characteristics of ram-jet engine at supersonic flight speeds.
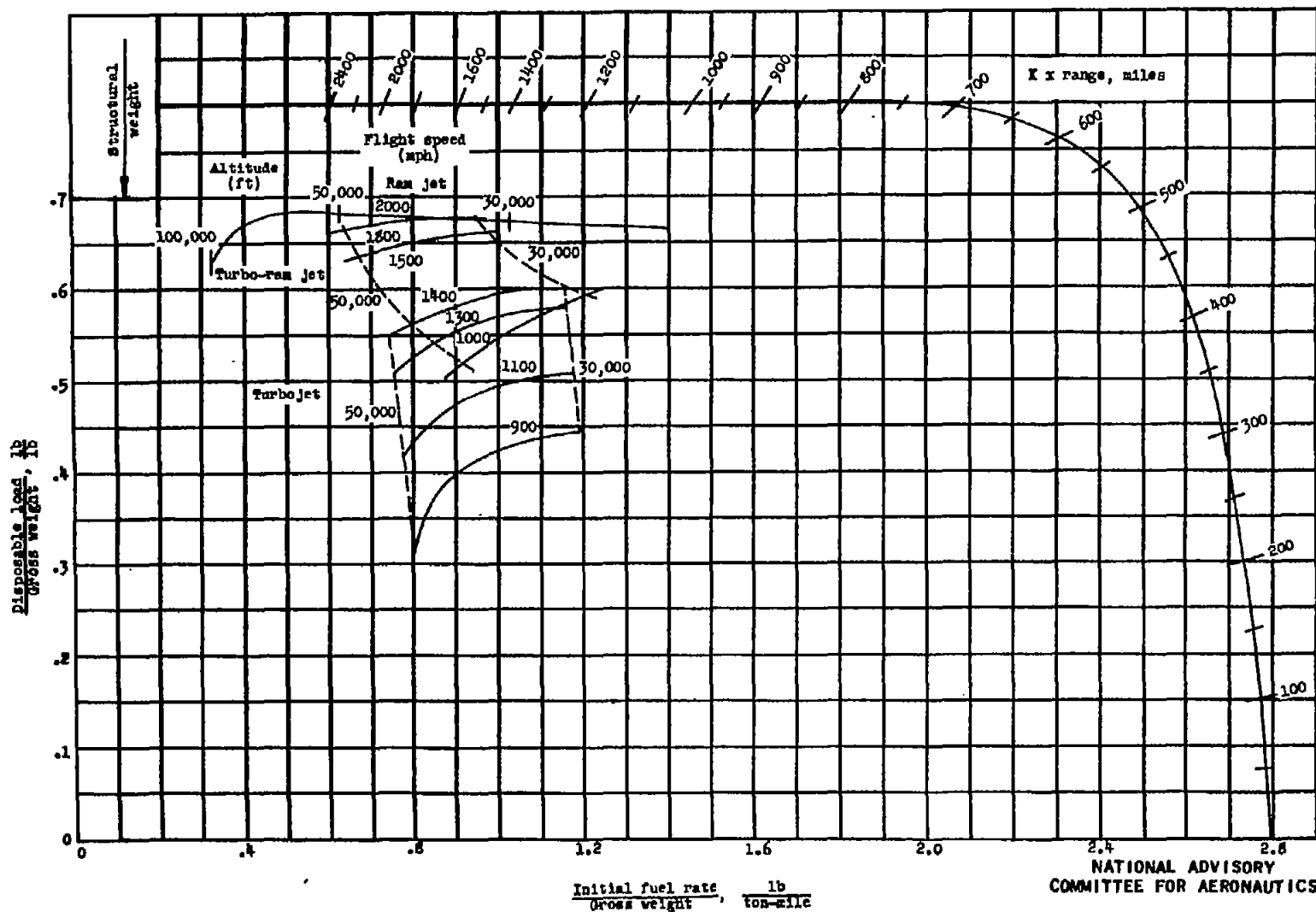
Figure V-6. — Comparison of load-range characteristics of turbojet, turbo-ram-jet, and ram-jet engines at supersonic flight speeds.

Fig. V-8

VI - THE ROCKET ENGINE

In this section of the report, the rocket power plant is briefly described, some of the performance parameters are discussed, and the use of this power plant in two of its many applications is considered. The first case to be considered is that of the rocket-powered projectile; the second case is the use of the rocket power plant in an airplane.

Description

The rocket-propulsion system is probably the oldest and simplest propulsion system recognized. The rocket carries oxidant in addition to fuel and thus has the unique characteristic of being entirely independent of the atmosphere for operation.

The rocket power plant comprises essentially a rocket engine, consisting of a combustion chamber and a nozzle, and a propellant-(fuel plus oxidant) supply system. The propellant-supply system may either be contained wholly within the combustion chamber (for example, as a solid material such as used in ordinary pyrotechnic skyrockets) or the system may consist of propellant tanks, valves, controls, injectors, and a pressurizing or a pumping system such as would be required for using liquid propellants. An example of a rocket power plant using liquid propellants, which is the type discussed in this part of the report, is the well-known German V-2 rocket power plant in which liquid oxygen and alcohol were supplied to the combustion chamber by means of high-pressure pumps.

Performance Parameters

The rocket-propulsion principle is diagrammatically illustrated in figure VI-1. In the combustion chamber of the rocket engine, the propellants react either spontaneously or after suitable ignition, releasing large amounts of heat energy and generating high-temperature gases at a high rate. By expanding the high-temperature gases through the nozzle, a portion of the heat energy liberated in the combustion chamber is converted into kinetic energy of flow. The reaction to the momentum of the ejected gases results in the thrust that propels the rocket, or

$$F = \frac{w_f}{g} u_e \qquad (1)$$

where

F   thrust, pounds

w_f mass rate flow of propellant, pounds per second

g   conversion factor, 32.2 pounds per slug

u_e effective exhaust velocity, feet per second

Theoretically the effective exhaust velocity differs from the axial velocity at the center of the nozzle exit by a factor that corrects for the angle of divergence α of the nozzle and a pressure correction term that allows for any difference existing between the exit and ambient pressures. For divergence angles below about 15° and for small differences between the nozzle exit and ambient pressures, the effective exhaust velocity is theoretically within a few percent of the axial velocity.

The specific impulse I, which is one of the primary rocket-engine performance parameters, is defined as

$$I = \frac{F}{w_f} \qquad (2)$$

and is, of course, equal to u_e/g.

The specific impulse is the reciprocal of the thrust specific propellant consumption, in units of seconds; therefore, for low values of specific propellant consumption, obviously high values of specific impulse are desired.

Equations (1) and (2) show that the thrust may be increased either by increasing the mass rate flow of propellant, which usually requires increasing the size of the rocket, or by increasing the effective exhaust velocity. The effective exhaust velocity or specific impulse is essentially a measure of the heat energy available for conversion into kinetic energy of flow and the efficiency of the conversion.

The theoretical relation for specific impulse derived on the basis of perfect gas laws and an isentropic expansion through the nozzle to ambient pressure is

$$I = \sqrt{\frac{R}{g} \frac{T_1}{M} \frac{2\gamma}{\gamma - 1} \left[1 - (p_3/p_1)^{\frac{\gamma-1}{\gamma}} \right]} \qquad (3)$$

where

R   universal gas constant, 1545 foot-pounds per pound-mole per $^\circ R$

T_1 combustion temperature, $^\circ R$

M   molecular weight of products of combustion

γ ratio of specific heats

p_1 combustion-chamber pressure, pounds per square inch

p_3 nozzle-exit pressure, pounds per square inch

Equation (3) indicates that to obtain high values of specific impulse the following properites would be desirable: high combustion temperatures, low molecular weight of the gases, high combustion-chamber pressures, low nozzle-exit pressures, and low ratios of specific heat. The effect of these factors on specific impulse are shown in figure VI-2. Values of the quantity

$$\Delta = \frac{2\gamma}{\gamma - 1} \left[1 - (p_3/p_1)^{\frac{\gamma-1}{\gamma}} \right]$$

are shown plotted against pressure ratio p_1/p_3 for several values of γ in figure VI-2(a). Values of the theoretical specific impulse I are shown as a function of T_1/M for several values of Δ. The value of Δ increases with pressure ratio but the rate of increase is greatly reduced at the high pressure ratios. The value of Δ also increases with decreasing values of γ. Appreciable increases in specific impulse can be realized by increasing the value of T_1/M and, of course, the specific impulse increases with increasing values of Δ.

The V-2 rocket engine operated with a chamber pressure of approximately 300 pounds per square inch (sea-level pressure ratio, 20) and a value of T_1/M of about 250° R resulting in a theoretical specific impulse of about 245 pounds-seconds per pound.

In figure VI-3, theoretical values of specific impulse I and
the product I_d of specific impulse and density of propellant are
compared in a bar graph for several of the well-known liquid propel-
lants at a pressure ratio of 20. The values of I represent thrust
per unit weight flow and the values of I_d represent thrust per
unit-volume flow. The comparison of the values of I_d is important
from the standpoint of the size of the propellant tanks required
and its effect on the weight and drag of the tanks. Thus, from this
consideration hydrogen-oxygen, with a theoretical value of I of
about 350 pounds-seconds per pound is probably not better than the
alcohol-oxygen mixture because of the low value of I_d for hydrogen-
oxygen.

Other factors, of course, have to be considered in the selection
of a rocket propellant, among which are availability, cost, handling,
and storage characteristics.

Rocket-Powered Projectile

The first application of the rocket power plant considered is
the rocket-powered projectile. By far the greatest part of the
range of a projectile, unlike an airplane, is covered in free flight
(coasting after the end of power). The calculation of the range of
a projectile involves a definite flight plan and a large number of
details. In order to illustrate the effect of some of the variables
on the maximum range of a projectile on the earth's surface, however,
the problem may be simplified by assuming a ballistic trajectory
(negligible burning time), and by neglecting the drag of the projec-
tile. Figure VI-4, based on these assumptions, shows values of specific
impulse I plotted against the velocity of the projectile at the end
of power for four ratios of the propellant weight to the gross weight
of the projectile. Included in this figure is an approximate scale of
the maximum range of the projectile on the earth's surface. Figure VI-4
shows that the range increases with about the square of the specific
impulse, that is, increasing the specific impulse by 50 percent practi-
cally doubles the range. This factor serves to illustrate further
the importance of increasing the specific impulse. The fact that the
range of the projectile varies with about the square of the specific
impulse is an essential difference between a projectile and the air-
plane to be considered, for which range varies with about the first
power of specific impulse.

Large increases in range can also be realized by increasing the
ratio of propellant weight to gross weight. A limit exists, however,
on the value of this ratio that can be attained with a single rocket.

The V-2 rocket with pay load, for example, had a propellant to gross
weight ratio of about 0.65. A possible method for increasing, in
effect, this ratio is the step-rocket in which two or more rockets are
joined as a unit. The rockets are arranged to burn consecutively and
each step is discarded when exhausted of power. With step-rockets
and available propellants, the velocity of the final step at the end
of power could be about 5 miles per second or 18,000 miles per hour,
which as indicated in figure VI-4 would permit flight to any point on
the earth's surface or would permit establishing a permanent orbit at
the earth's surface. With a velocity of approximately 7 miles per
second or 25,000 miles per hour, the final step would escape from
the earth's gravitational field.

Rocket-Powered Airplane

In the second case of the application of the rocket power plant,
a rocket-powered airplane is considered. The general assumptions
made concerning the airplane in the previous parts of this report
and listed in the appendix were followed. The V-2 rocket engine and
the following actual available data (reference 2) for this engine
were used:

Specific impulse, pounds-seconds per pound 218
Sea-level thrust, pounds 60,000
Engine weight, pounds 2235
Maximum engine diameter, feet 3.11

In figure VI-5, the thrust per unit engine weight is shown
plotted against altitude for the V-2 engine for the actual specific
impulse of 218 pounds-seconds per pound and a curve for a specific
impulse of 300 pounds-seconds per pound is included for comparison.
At a given altitude the thrust of the rocket engine, unlike the
engines discussed in the previous parts of this report, is essentially
constant and independent of flight speed. The thrust increases slightly
with altitude as a result of the free expansion of the gases from the
exit pressure to the lower ambient pressure at altitude. The thrust
per unit engine weight ranges from about 27 to 31 pounds per pound
for a specific impulse of 218 pounds-seconds per pound. These values
compare with the following approximate values for the engines dis-
cussed in the previous parts of this report operating at conditions
for best range:

Engine	Thrust per unit engine weight (lb/lb)
Compound	0.6
Turbine propeller	.7
Turbojet	2.3
Turbo-ram jet	7.5
Ram jet	8.5

It is thus apparent that the thrust per unit engine weight for the rocket is appreciably higher than that for any of the other engines.

By increasing the specific impulse by approximately 37 percent, up to a value of 300 pounds-seconds per pound, the thrust and the thrust per unit engine weight of the rocket would be increased by an equal percentage.

The thrust per unit engine frontal area and the thrust specific propellant consumption for the rocket engine are plotted in figure VI-6 as a function of altitude for specific impulses of 218 and 300 pounds-seconds per pound. The thrust per unit engine frontal area (fig. VI-6(a)) for a specific impulse of 218 pounds-seconds per pound ranges from about 8000 to 9000 pounds per square foot, as compared with the following values for the engines discussed in the previous parts of this report operating at conditions for best range:

Engine	Thrust per unit engine frontal area (lb/sq ft)
Compound	230
Turbine propeller	265
Turbojet	415
Turbo-ram jet	1800
Ram jet	2000

The appreciably larger value of thrust per unit engine frontal area for the rocket is indicative of its compactness. This compactness, the large thrust per unit engine weight, and the simplicity of the rocket engine constitute some of the primary advantages of the rocket. These advantages, however, are obtained at the cost of a relatively high thrust specific propellant consumption, as shown in figure VI-6(b) because the rocket carries its entire working mass. The

thrust specific propellant consumption shown for the V-2 engine
ranges from approximately 16.5 at sea level to 14.5 pounds per hour
per pound at the higher altitudes, compared with the following values
for the other engines operating at conditions for best range:

Engine	Thrust specific fuel consumption $(lb/(hr)(lb))$
Compound	0.22
Turbine propeller	.26
Turbojet	1.6
Turbo-ram jet	2.2
Ram jet	2.0

The load-range characteristics of the rocket-powered airplane
are shown in figure VI-7. The disposable load in pounds per pound
gross airplane weight is plotted against the initial propellant rate
in pounds per mile per ton gross airplane weight for several constant
altitudes and supersonic flight speeds. Also included in this figure
are a K × range   scale and broken lines indicating the location of
the origin for convenience in estimating the range. This range,
unlike that for the projectile is, of course, only for the powered
flight.

Inasmuch as the diameter of the engine was less than that of the
propellant tanks (fuselage), it was assumed that the engine could be
placed in rear of the fuselage and hence, the drag of the engine was
taken as zero. In addition, only the wave drag of the front of the
fuselage was used in calculating wave drag.

At sea level, increasing the flight speed rapidly increases the
initial propellant rate and consequently decreases the range. At an
altitude of 100,000 feet, however, increasing the flight speed decreases
the initial propellant rate and hence increases the range. At altitudes
between these values there is a transition in the effect of speed on
range. These differences in the effect of speed on range occur because
the predominant drag is from the fuselage at low altitudes and from
the wings at high altitudes. For the conditions presented, the change
in the disposable load, as for the ram jet, is small compared with the
other engines.

At an altitude of 50,000 feet and a flight speed of 2000 miles
per hour, a gross airplane weight of about 365,000 pounds was obtained.
In order to illustrate the effect of gross weight on range this con-
dition was recalculated to give gross weights of approximately 200,000
and 50,000 pounds by assumming that the weight and the thrust of the

engine varied with the square and the cube, respectively, of the
engine diameter. The points are indicated by symbols in figure VI-7.
It is shown that by changing the gross weight from 200,000 to
50,000 pounds the disposable load is changed by a negligible amount
and the range is decreased by about 15 percent.

The following table lists the approximate values of gross air-
plane weight for the various altitude and flight-speed conditions
considered:

Altitude (ft)	0	30,000	50,000	100,000
Flight speed (mph)	Gross weight (lb)			
1000	234,500	377,600	440,200	482,500
2000	98,300	249,000	365,600	474,200
2000	-------	-------	200,000[a]	-------
2000	-------	-------	50,000[b]	-------
3000	46,700	158,900	292,000	462,900
5000	17,500	70,000	177,300	434,900

[a]Engine thrust, 40,000 lb; engine weight, 1550 lb.

[b]Engine thrust, 11,200 lb; engine weight, 670 lb.

The best operating condition shown for the rocket is a flight
speed of 5000 miles per hour and an altitude of 100,000 feet. At
this condition, the disposable load is 0.695 pound per pound gross
weight, the initial propellant rate is 0.91 pound per ton-mile gross
weight, and the indicated range is 1387 miles

Comparison and Application

For purposes of comparing the rocket engine with the jet engines
discussed in the previous parts of this report, conditions at a flight
speed of 3000 miles per hour and altitudes of 50,000 and 100,000 feet
were chosen for the rocket engine. The comparison is shown in figure VI-8
for the rocket, the ram-jet, the turbo-ram-jet, and the turbojet engines
in a plot similar to that of figure VI-7. The disposable load is slightly
higher for the rocket than for the ram jet but the initial propellant
rate has been increased with a consequent decrease in range. The rocket
engine therefore would have applications in high-speed, short-range

airplanes where low engine weight, compactness, and simplicity of the engine are at a premium and propellant consumption is a secondary consideration.

In addition to the ability of the rocket engine to provide propulsion outside the earth's atmosphere, the rocket is unique in providing enormous amounts of thrust from a simple and compact unit, and thus is applicable in cases such as powering artillery-type projectiles, missiles such as the V-2, and auxiliary power for airplanes, pilotless aircraft, and missiles.

Fig. VI-1

Combustion
chamber

F

Throat   Exit

$$F = \frac{w_f}{g} u_e$$

u_e

a
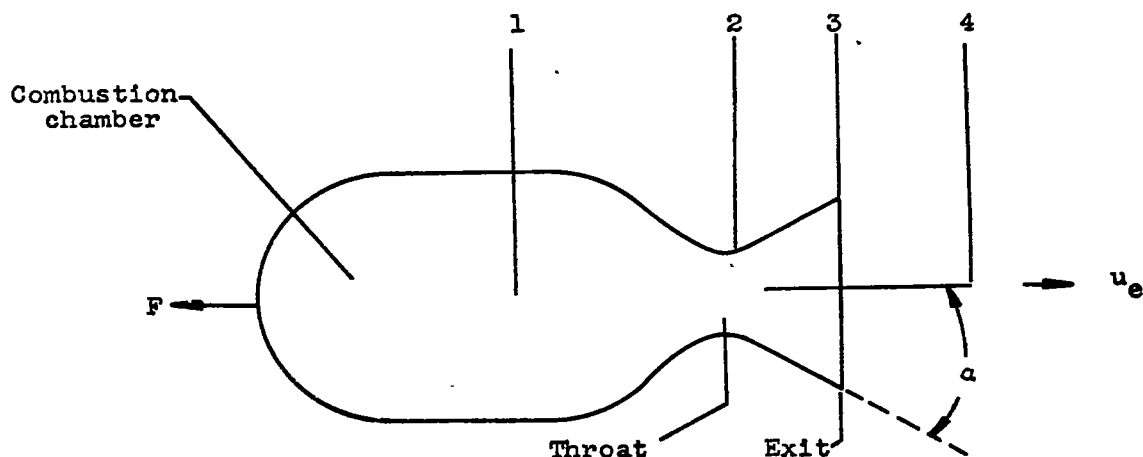
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Figure VI-1. — Diagrammatic sketch of the rocket-propulsion
principle.

Fig. VI-2   NACA TN No. 1349

$$\Delta = \frac{2\gamma}{\gamma-1} \left[1 - (p_3/p_1)^{\frac{\gamma-1}{\gamma}} \right]$$

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γ

1.1
1.2
1.4
1.6

p_1/p_3
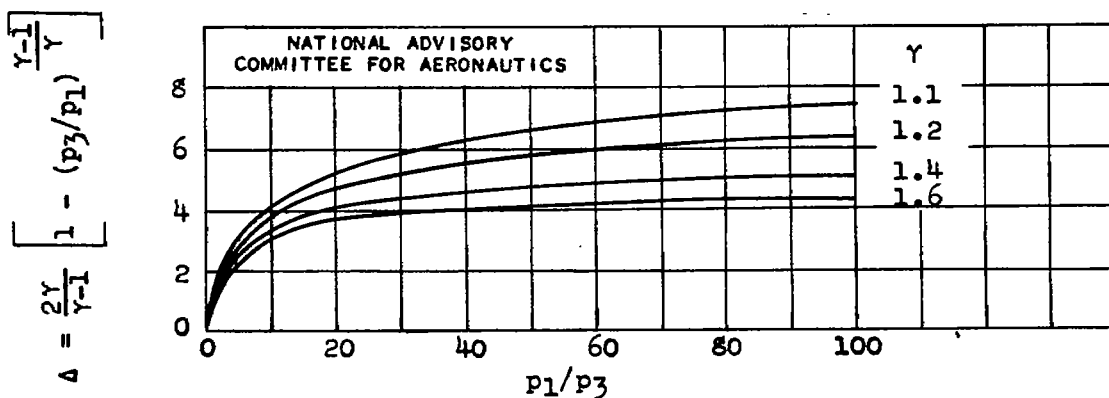
(a)   Variation of Δ with pressure ratio for several
values of γ.

$$I = \sqrt{\frac{R}{g} \frac{T_1}{M} \frac{2\gamma}{\gamma-1} \left[1 - (p_3/p_1)^{\frac{\gamma-1}{\gamma}} \right]}$$

Specific impulse, I, lb-sec/lb

Δ

8
6
4
2

T_1/M, °R
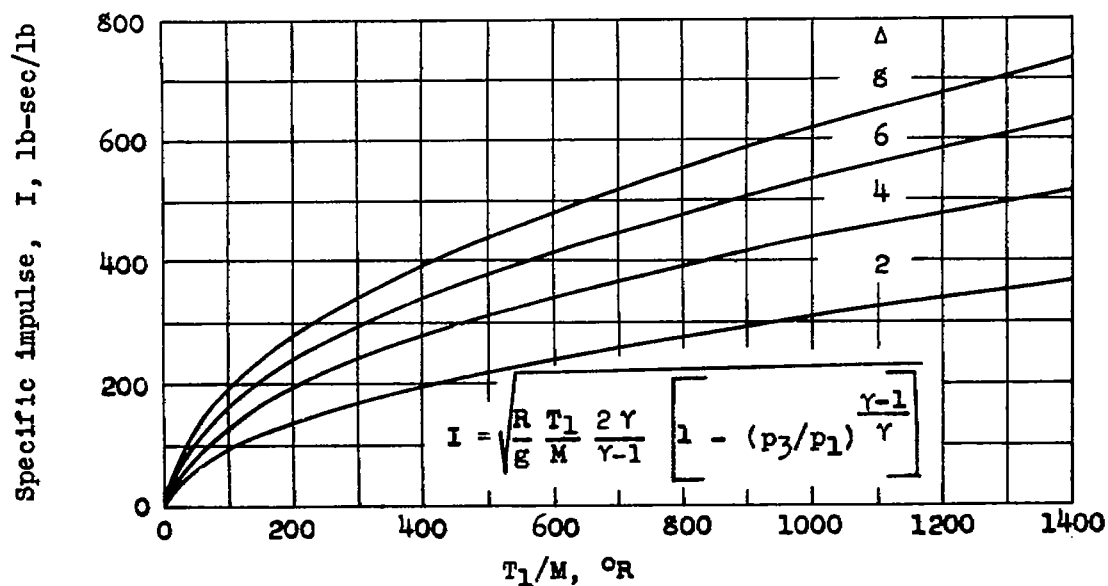
(b)   Variation of specific impulse with T_1/M for
several values of Δ.

Figure VI-2. — Specific impulse, a function of the propellant system.

LIQUIDS

Hydrogen + oxygen

Ethyl alcohol + oxygen

I

$I_d/62.4$

Methyl alcohol + hydrogen peroxide

Aniline + nitric acid

0   100   200   300   400

I, lb-sec/lb   and $I_d/62.4$, lb-sec/cu ft
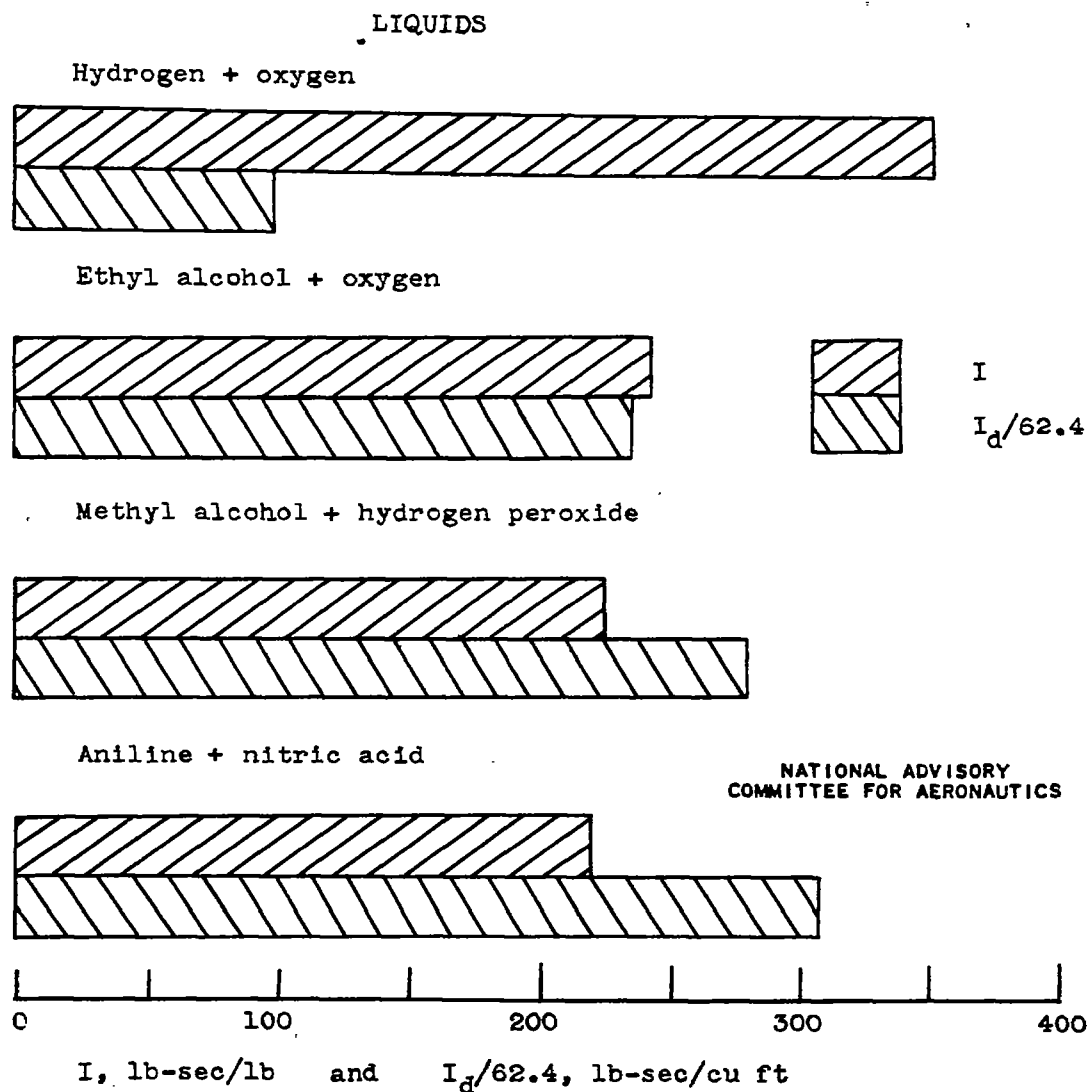
Figure VI-3. — Comparison of theoretical values of specific impulse
I and product I_d of specific impulse and density of propellant
for several liquid propellants. Pressure ratio, 20.

Fig. VI-4   NACA TN No. 1349

Maximum range on earth's surface, miles
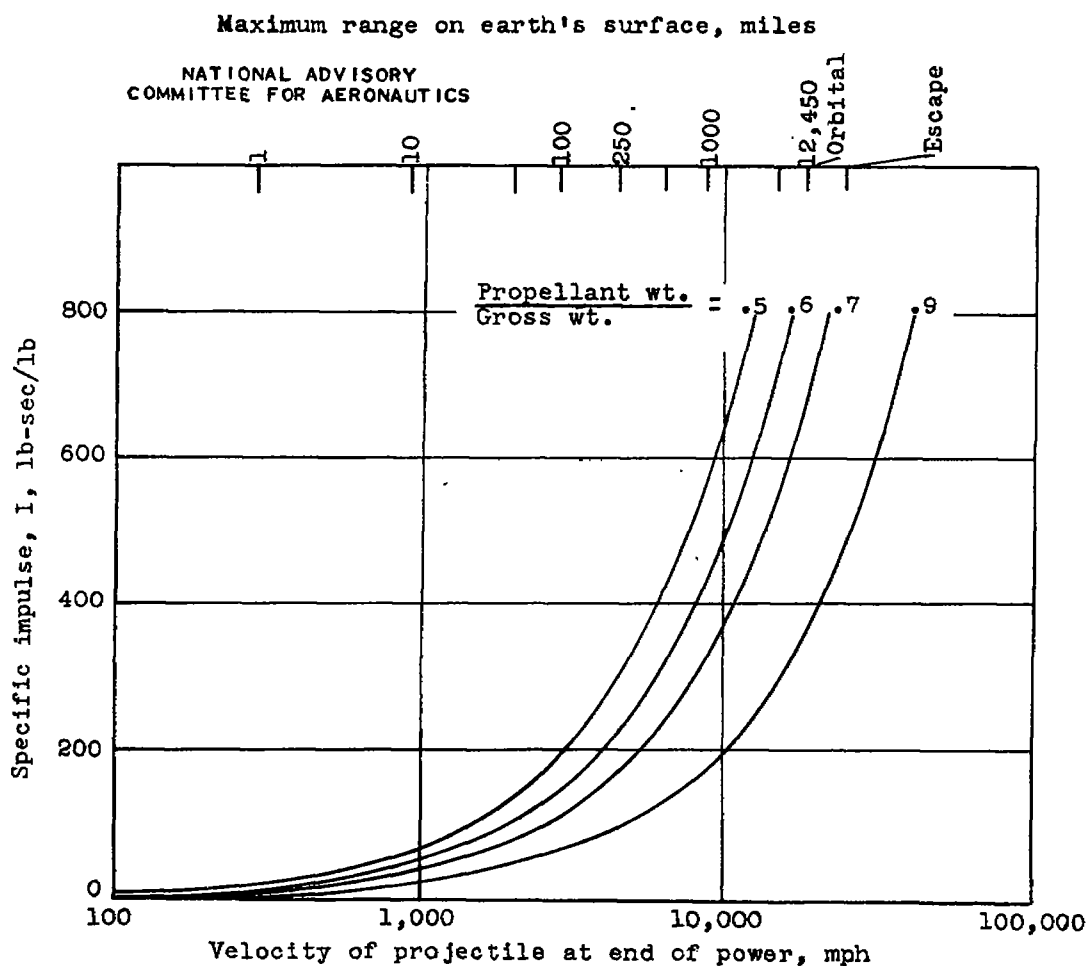
Figure VI-4. - Variation of velocity at end of power and corresponding
range of a rocket-powered projectile with specific impulse and ratio
of propellant weight to gross weight. Ballistic trajectory; drag
neglected.
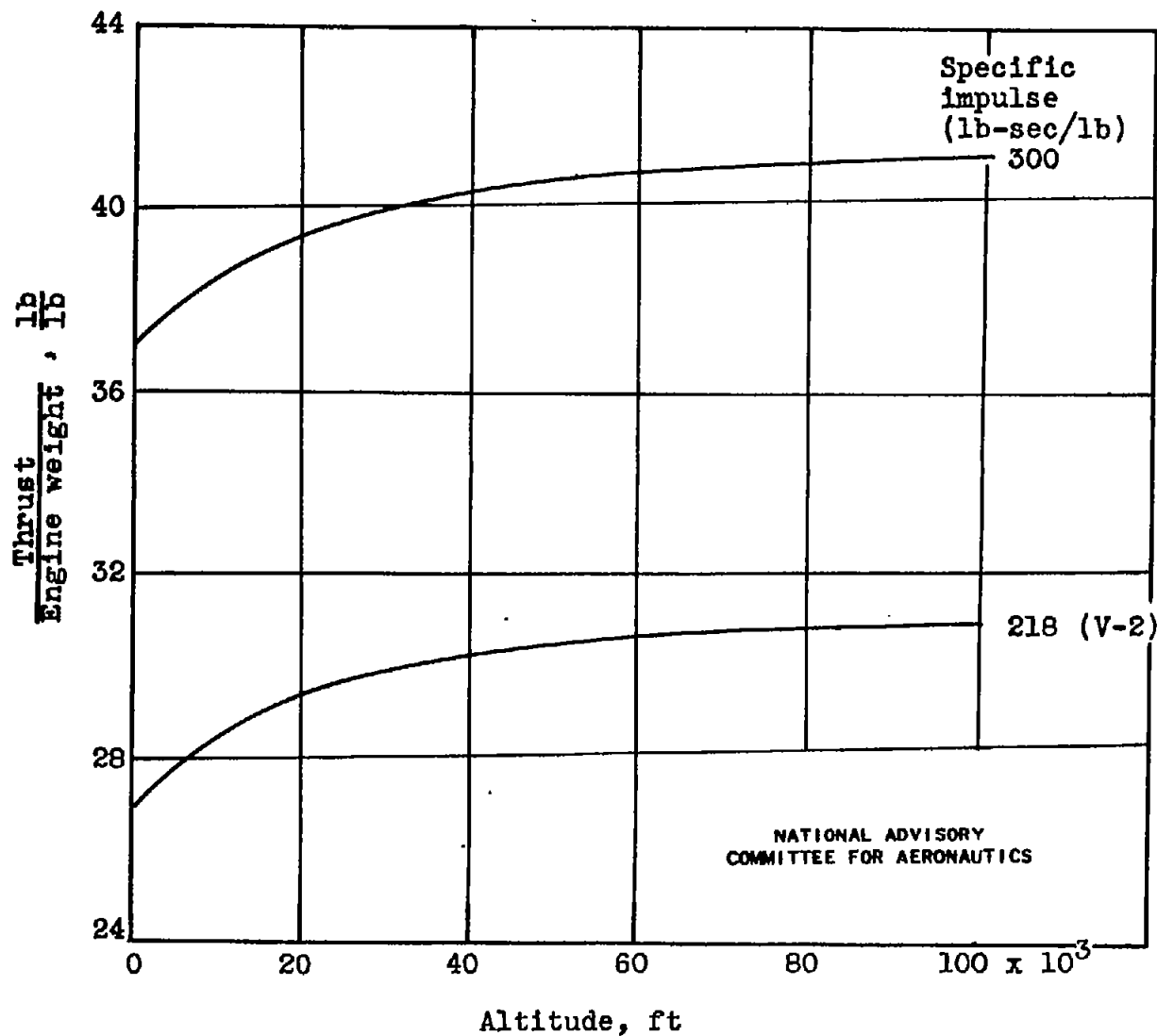
Figure VI-5. - Variation of thrust per unit engine weight with altitude for two values of specific impulse.

Fig. VI-5

Fig. VI-6   NACA TN No. 1349
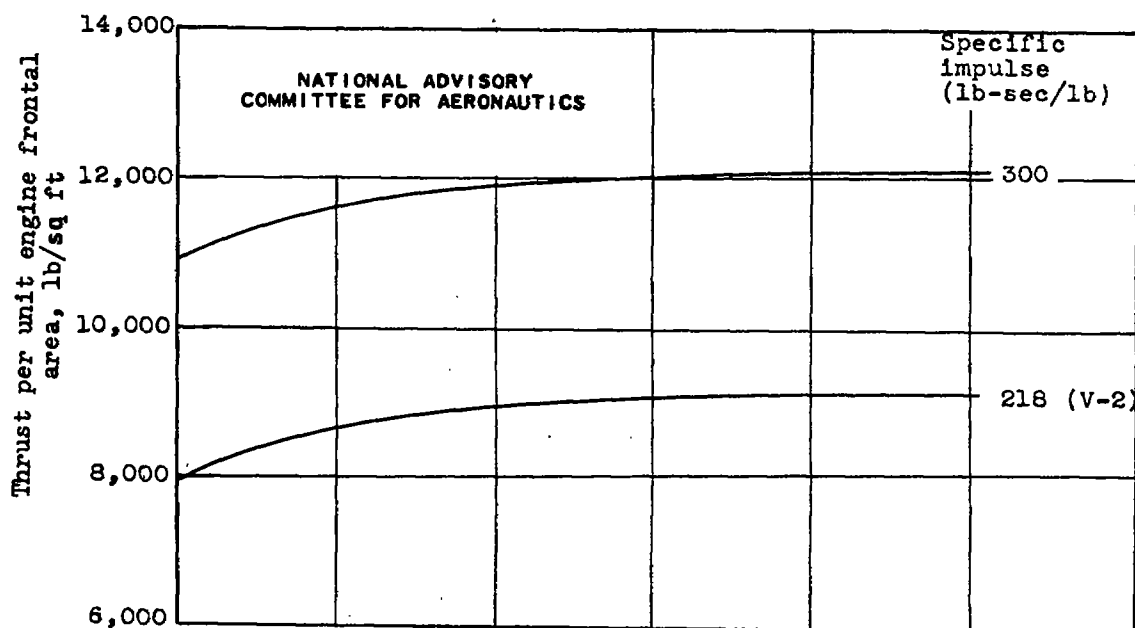
(a)   Variation of thrust per unit engine frontal area with altitude.
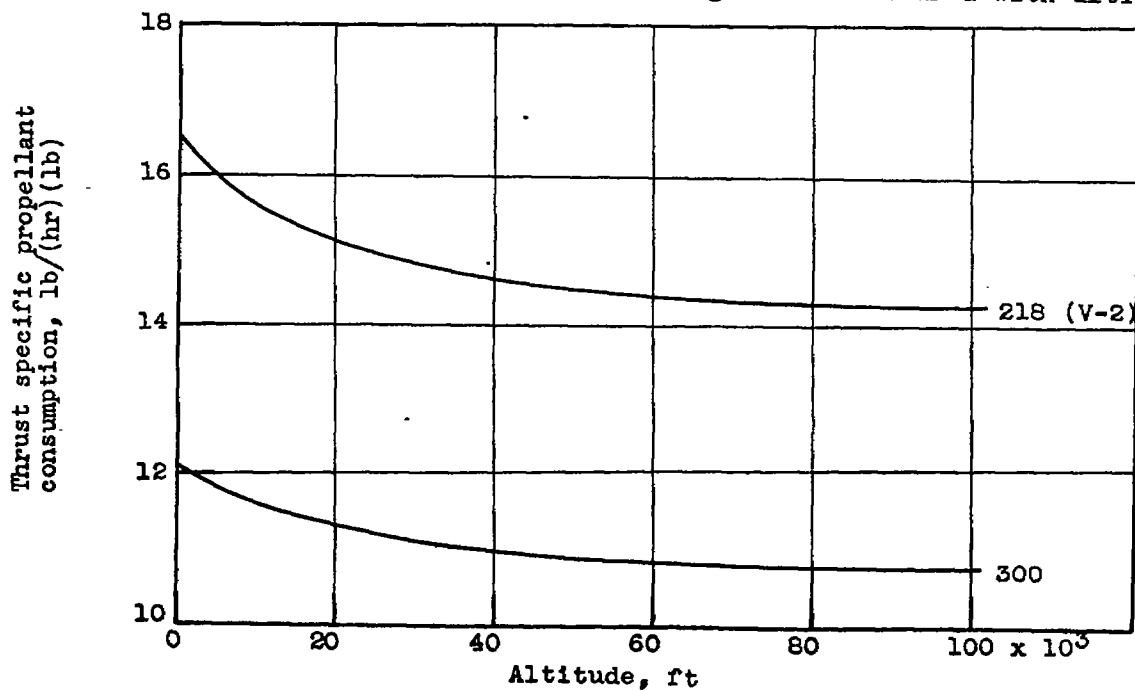
(b)   Variation of thrust specific propellant consumption with altitude.

Figure VI-6. – Variation of thrust per unit engine frontal area and thrust specific propellant consumption with altitude for two values of specific impulse.
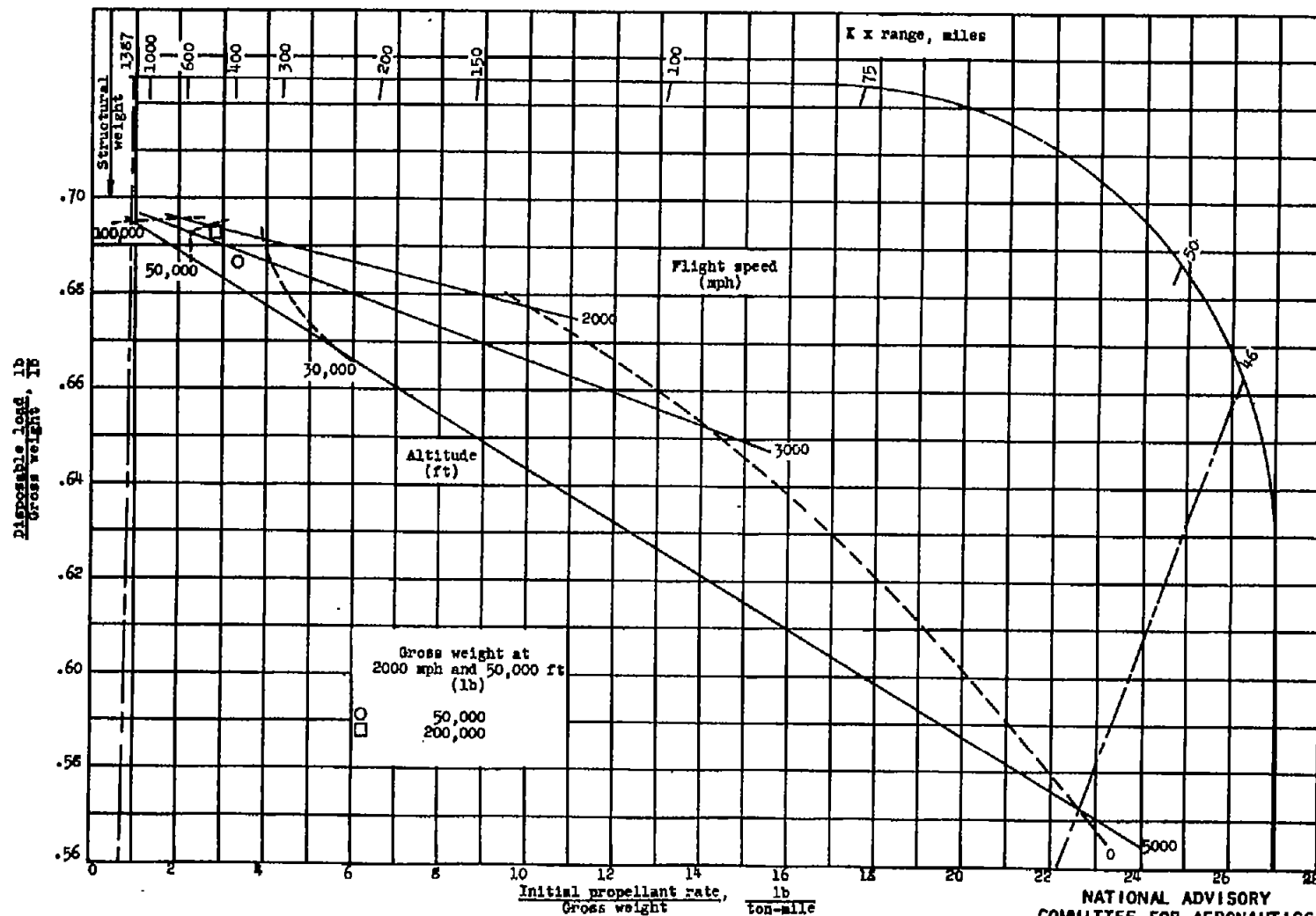
Figure VI-7. — Load-range characteristics of rocket engine at supersonic speeds.

Fig. VI-7

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Fig. VI-8

K x range, miles

Flight speed (mph)

Altitude (ft)

Ram jet

Rocket

Structural weight

50,000

30,000   2000   3000

1500   100,000   50,000

100,000   1500

Turbo-ram jet

1000

1000

1300

Flight speed (mph)

1100

900
50,000

30,000

Turbojet

30,000

2400 2000 1600 1400 1200 1000 900 800 700 600 500 400 300 200 100

.8

.7

.6

.5

.4

.3

.2

.1

0

Disposable load, lb / Gross weight, lb

0   .2   .4   .6   .8   1.0   1.2   1.4   1.6   1.8   2.0   2.2   2.4   2.6   2.8

Initial fuel rate (propellant for rocket) / Gross weight , lb/ton-mile

Figure VI-8. - Comparison of load-range characteristics of rocket, ram-jet, turbo-ram-jet, and turbojet engines at supersonic flight speeds.

DISCUSSION OF POSITION OF SIX AIRCRAFT-PROPULSION

SYSTEMS IN SPEED-RANGE SPECTRUM

The thrust per unit engine frontal area, thrust per unit engine weight, specific fuel consumption, load-carrying capacity, fuel-consumption rate per ton-mile, and range for the various power plants analyzed are discussed in detail in the individual parts of this report. In this section a recapitulation is made of the performance of the various propulsion systems on the basis of their position in plots of disposable load against fuel rate per ton-mile and range. Each point in these plots is a design point, that is, the engine is assumed to be designed specifically for the operating conditions corresponding to that point.
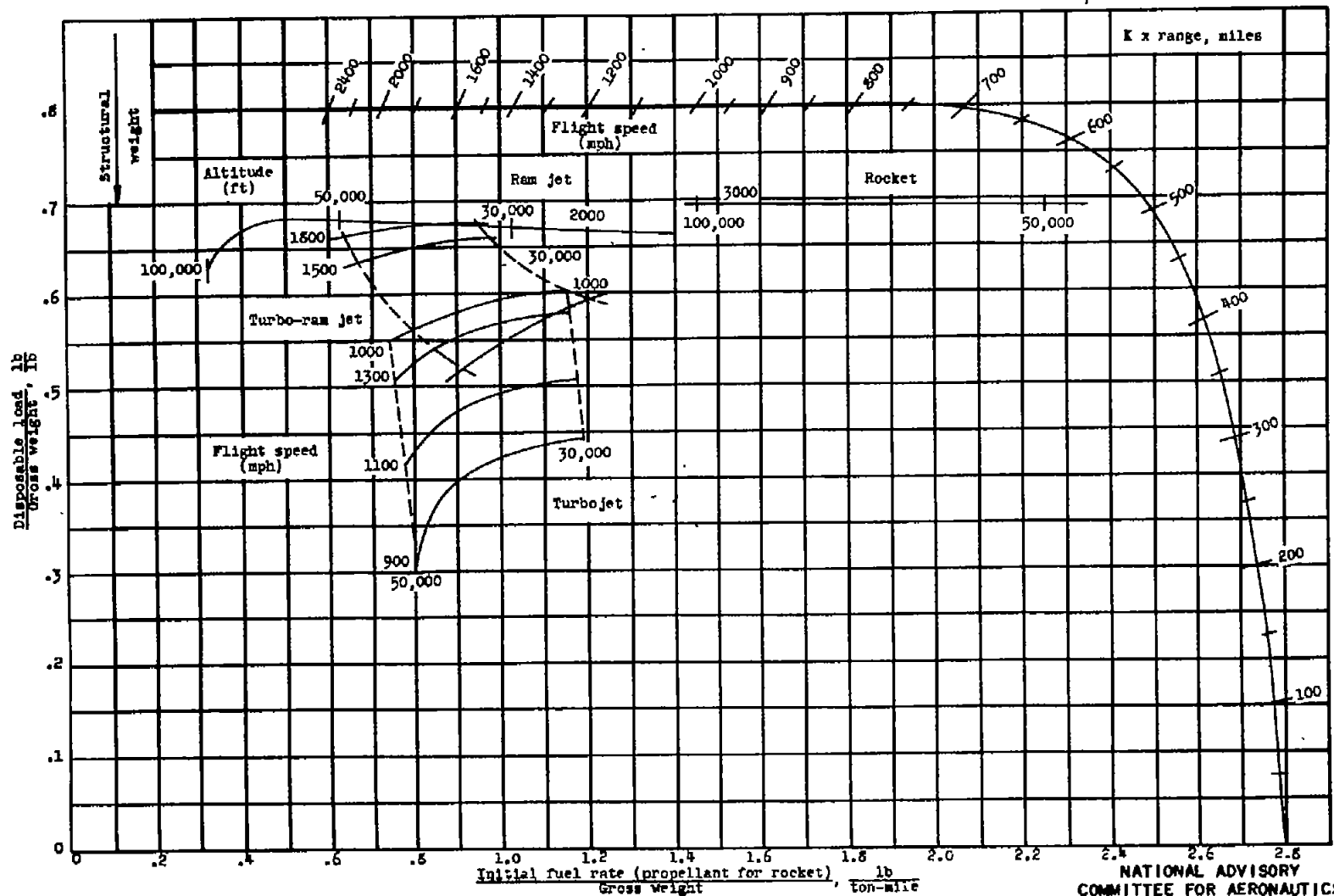
The disposable load per pound of airplane gross weight is plotted against initial fuel rate per ton-mile (based on gross weight of airplane) in figure D-1 for subsonic flight for two cases:

(a) Constant lift-drag ratio condition: A constant lift-drag ratio of 18 for the airplane (minus nacelle) was assumed at all flight conditions (fig. D-1(a)).

(b) Limiting wing-loading condition: A lift-drag ratio of 18 was assumed only for flight conditions for which the resulting wing loading is 80 pounds per square foot or less; for other flight conditions, the value of the lift-drag ratio was reduced to give a wing-loading value of 80 pounds per square foot (fig. D-1(b)). The values of lift-drag ratio for this case are shown in figure D-2.

The disposable load in figure D-1 includes the weight of fuel plus tanks and the pay-load weight. The values on the $K \times$ range scale shown were obtained by computing the ratio of the disposable load to the initial fuel rate per ton-mile multiplied by a factor of 2000 pounds per ton to correct for the difference in units between the ordinate and abscissa and divided by 1.1 to adjust the range for the weight of fuel tanks. The factor K corrects for the variation in fuel rate per ton-mile during the flight. It is defined as the ratio of the average to the initial fuel rate per mile per ton of initial gross weight. The value of K depends on the flight plan and the gross weight of the airplane at the start and at the end of flight. It may be computed for a large number of flight plans by means of equation (A9) of the appendix. Illustrative values of K, determined from the Breguet range equation (equation (A13)) in which it is assumed that the flight is made at a constant lift-drag ratio and constant specific fuel consumption (lb/hp-hr), are shown by the curve on the left side of figure D-1. From the abscissa of this graph, the

value of K corresponding to a value of the ordinate equal to the ratio of the fuel load consumed in flight to the initial gross weight of the airplane can be read.

Illustrative curves are shown for the compound, the turbine-propeller, the turbojet, and the turbo-ram-jet engines. The curves for the turbine-propeller and compound engines overlapped and to avoid confusion the parts of the curves of each engine were deleted in the region where that engine gave lower disposable load than the other for the same flight speed and fuel rate per ton-mile. The maximum range at each operating point is obtained when the total disposable load is assumed to be fuel. The value of $K \times$ range corresponding to this condition is obtained by drawing a straight line through the origin and the desired flight condition to the $K \times$ range scale. Such a line is illustrated in figure D-1(a) for the compound engine at a flight speed of 200 miles per hour and an altitude of 30,000 feet. It is noted that for these conditions the value of $K \times$ range is approximately 7400 miles. By reading horizontally from the value of the ratio of disposable load to gross weight, a value of K of 0.74 is obtained from the curve on the left side of the figure. When this value of K is divided into $K \times$ range, it gives the value of range of 10,000 miles. For a shorter range it is possible to carry a pay load and the division between fuel and pay load can be read from this figure. For example, for a value of $K \times$ range of 2000 miles, as illustrated in figure D-1(a), the vertical distance from the point corresponding to a desired flight condition to the line that connects $K \times$ range = 2000 to the origin is the pay load per pound of gross weight, and the remainder of the vertical distance to the abscissa is the fuel load (including tank) in pounds per pound of gross weight. The value of K is determined from the left-hand curve corresponding to this value of fuel load (including tank) per airplane gross weight. Fuel reserve for emergency must, of course, be deducted from the pay load.

The structural weight and the engine weight per unit of initial gross weight of the airplane can also be read from figure D-1. The distance from unity to 0.6 represents the structural weight per unit gross weight because it was assumed in the preparation of this chart that the structural weight per unit gross weight was 0.4. The vertical distance from the structural weight line (0.6 ordinate in fig. D-1) to any desired operating point gives the value of the installed engine weight (including propeller for the propeller-type engines) per unit of airplane gross weight.

The effect of a change in the assumptions with regard to the structural weight or the engine weight can readily be seen in this

figure. For example, if the structural weight per unit gross weight
is decreased from 0.4 to 0.3, the improvement is directly reflected
as an equal increase in disposable load, that is, each point in fig-
ure D-1 is raised by an amount of 0.1.

The effect of a reduction in engine weight is introduced by
reducing the vertical distance from the operating point to the struc-
tural weight line (0.6 in fig. D-1) by an amount equal to the per-
centage reduction in installed engine weight. It is noted that for
the compound engine and the turbine-propeller engine a reduction in
weight at high flight speeds has a greater beneficial effect on the
range of the airplane than the same percentage reduction in engine
weight at a low flight speed. The effect of a 40-percent reduction
in weight of the turbine-propeller engine (including propeller) at
a flight speed of 500 miles per hour is illustrated in figure D-1(b)
by the dashed curve labeled A, which was obtained by moving the
solid curve for 500 miles per hour for this engine vertically in the
manner just described.

The effect of a change in specific fuel consumption from the
values used in the preparation of figure D-1 can be introduced by
changing the abscissa values proportionally to the change in specific
fuel consumptions. The values of engine weight per unit thrust and
specific fuel consumptions used in the preparation of the summary
figures can be obtained from the individual parts of this report.

The effect of a change in lift-drag ratio L/D (airplane minus
nacelles) can be indicated in figure D-1 for any given operating
point by moving the point along a line passing through the operating
point and point X (located at the coordinates abscissa = 0, ordin-
ate = 0.6) on the basis that the distance of the operating point
from point X is inversely proportional to the value of L/D. The
points in figures D-1(a) and D-1(b) at the same operating condition
for a given engine therefore fall on a common line passing through
point X.

The effect of change in the ratio r of nacelle drag to engine
thrust can be indicated in a similar manner on the basis that the
distance from the operating point to point X is inversely propor-
tional to 1 - r. For example, the effect of shifting to a completely
submerged installation (r = 0) can be obtained by moving the oper-
ating point in figure D-1 toward point X a distance proportional to
the corresponding value of r used in the preparation of figure D-1.
The values of r corresponding to the operating conditions of fig-
ure D-1 can be obtained from the curves in the individual parts of
this report. Points B, C, and D in figure D-1(b) were obtained in the

manner just described and illustrate the effect of a shift to a completely submerged installation for the compound, turbojet, and turbine-propeller engines, respectively, for the highest speed shown for each of these engines at the best altitude, as shown in the following table:

Point	Engine type	Speed, mph	Altitude, ft
B	Compound	500	40,000
C	Turbine propeller	500	30,000
D	Turbojet	550	50,000

The importance of submerging the engine for high-speed flight is evident.

Examination of equations (A5) and (A7) reveals the basis for the foregoing discussion on shifting the position of the curves in figure D-1.

For high flight speeds at low altitude, the condition of a constant lift-drag ratio L/D of 18 (fig. D-1(a)) imposes wing loadings far above the values currently used. The limited wing-loading condition (fig. D-1(b)) is in some respects the more practical condition in that it takes cognizance of the take-off and landing problems. It should be noted, however, that the performance shown in figure D-1(a) for the constant lift-drag ratio considered is possible even in the high wing-loading range if special means are provided for take-off, such as take-off from a mother ship at high speeds.

When flight speed is decreased or altitude is increased, a region of operation is reached where the lift-drag ratio of 18 can be obtained with a wing loading of 80 pounds per square foot or less. In this region the curves of figure D-1(b) agree with the corresponding curves of figure D-1(a). Outside of this region in the case of figure D-1(b), the lift-drag ratio must be reduced to meet the wing-loading condition with the result that a decrease in disposable load and increase in fuel rate per ton-mile is obtained with respect to the corresponding flight condition in figure D-1(a). The variation of the lift-drag ratio to meet the limiting wing-loading condition is shown in figure D-2. The adverse effect of the wing-loading limitation increases with increase in speed and decrease in altitude. Hence, with the wing-loading limitation it is necessary to fly at high altitudes to achieve long range at high speeds.

It is noted that of the engines considered the compound engine provides the lowest fuel rate per ton-mile. The cruise performance for this engine (engine speed, 2200 rpm; inlet-manifold pressure, 40 in. Hg absolute; fuel-air ratio, 0.063) was used in the preparation of figure D-1. The greatest range shown in figure D-1 for the compound

engine occurs at moderate-to-high altitudes and at the lowest speed
investigated (200 mph). The disposable load of the compound engine
decreases and the fuel rate per ton-mile increases as the flight
speed is increased. For high flight speeds, it is therefore neces-
sary to replace the compound engine by a lighter and smaller
frontal-area engine per unit thrust in order to restore the dis-
posable load to a high value. In the moderate altitude range (that
is, between 15,000 and 30,000 ft depending on speed), the turbine-
propeller engine shows better performance than the compound engine
at all speeds (fig. D-1). This superiority in performance is the
result of the lower weight and smaller frontal area per unit thrust
of the turbine-propeller engine at these altitudes.

Because of the reduction in power of the turbine-propeller
engine with increase in altitude, in contrast to the compound engine
(which is supercharged), the weight and the frontal area of the
turbine-propeller engine per unit power exceed that of the compound
engine at high altitudes. At high speeds it is advantageous to
operate at high altitudes in order to obtain a high lift-drag ratio
of the airplane consistent with a limited wing loading (fig. D-2).
Hence, because it maintains its power to high altitude, the com-
pound engine is capable of greater range than the turbine-propeller
engine at high speeds. (Compare for example, the curves for 500 mph
for the compound engine and turbine-propeller engine in fig. D-1(b)).

This analysis is based on a consideration of the weights and
performance of current engines and propellers. The turbine-propeller
engine is of recent development and large reductions in weight
per unit thrust may be achieved in the future. Special propellers
may be developed that will provide higher efficiency at high speed
than the propeller used in this analysis. When these improvements
are realized the turbine-propeller engine may be suitable for
much higher speed operation than indicated in the present analysis.
For example, the dashed curve in figure D-1(b) labeled A shows
the performance that may be obtained at 500 miles per hour if the
weight of the turbine-propeller engine (including propeller) is
reduced 40 percent. This analysis is limited to a discussion of
the engines on the basis of present performance and weights, and
no attempt is made to predict such future possibilities.

For both engines utilizing propellers, the disposable load per
ton of gross weight decreases rapidly and the fuel rate per ton-
mile increases with an increase in flight speed. The fuel rate per
ton-mile for the turbojet engine, however, decreases with increase
in flight speed because of the attendant increase in propulsive
efficiency. For very high speeds (550 mph and higher), the range
with the turbojet engine for the limiting wing-loading condition

(fig. D-1(b)) exceeds that for the engines utilizing propellers. As
compared with the compound and turbine-propeller engines, the weight
and the frontal area per unit thrust of the turbojet engine is low;
hence, the disposable load per unit gross airplane weight is high.
For short range operation at high speed, the load-carrying capacity
of the turbojet engine is therefore greater than for the two engines
using propellers.

In the case of the turbojet engine with constant lift-drag ratio
(fig. D-1(a)), the range is nearly independent of altitude; whereas
the range decreases rapidly with decrease in altitude with the wing-
loading limitation 80 pounds per square foot (fig. D-1(b)). For a
constant altitude of 30,000 feet, it is noted in figure D-1(b) that
as speed is increased the range first increases, reaches a maximum
at a flight speed slightly greater than 400 miles per hour, and then
decreases with further increase in flight speed. The increase in
range with flight speed up to the maximum range is the result of the
increased propulsive efficiency of the jet engine with speed. Maxi-
mum range occurs at the point at which the reduction in lift-drag
ratio introduced by the wing-loading limitation offsets the increase
in propulsive efficiency. The reduction in range with increase in
speed beyond this point is the result of the further reduction in
the lift-drag ratio required to meet the limiting wing-loading con-
dition. At an altitude of 50,000 feet, the lift-drag ratio of 18
does not result in the wing-loading limitation being exceeded at any
speed over the range shown and there is a progressive increase in
range with increase in speed.

The turbo-ram-jet engine provides a small increase in disposable
load with respect to the turbojet engine again at the cost of an
appreciable increase in fuel rate per ton-mile, with the result that
the maximum range is less than that of the turbojet engine. The
turbo-ram-jet engine can be converted to a turbojet engine merely by
shutting off the fuel flow to the tail pipe and adjusting the exit-
nozzle area. The tail-pipe burner can be turned on when boost power
is desired. The turbo-ram-jet engine shows a decrease in fuel rate
per ton-mile and an increase of range with increased flight speed.

The results of the analysis at supersonic speeds for a turbojet
engine, a turbo-ram-jet engine, a ram-jet engine, and a rocket engine
are summarized in figure D-3. In the case of supersonic flight, the
frontal area was kept at a minimum. The fuselage volume was taken
as that sufficient to house the disposable load on the assumption
that the entire disposable load has the denisty of fuel; the drag of
the fuselage was computed in each case on this basis. The rocket
engine was assumed to be located in the rear of the fuselage; whereas
the other engines were assumed to be housed in separate nacelles in

the wing and the drag of the nacelles was taken into account. The
wing was assumed to have a constant lift-drag ratio of 7. The ratio
of the structural weight to gross weight was taken as 0.3.

The maximum range, as in the previous figures, is obtained by
drawing a straight line from the origin through the operational
point desired to the K × range scale. The value of K for the
supersonic airplanes is close to unity (between 0.9 and 1.0 for
most points in fig. D-3) because the wing drag in the range of con-
ditions shown is small compared with the sum of the fuselage and
nacelle drag; hence, there is little change in drag of the airplane
with consumption of fuel and little change in the fuel rate per
mile experienced during a given flight. For any range less than the
maximum range, the pay load can be determined in the manner described
in the discussion of figure D-1. The structural weight and the
engine weight can be read from figure D-3 in the manner described
for figure D-1. The remarks on the determination of the effects of
variation of structural weight and engine weight made in connection
with figure D-1 are approximately true for figure D-3.

For the turbojet engine, the fuel rate per ton-mile decreases
with increase in altitude, but does not change greatly with speed
for a constant altitude for the range of conditions shown in fig-
ure D-3. The disposable load, however, increases with speed. For
each point in figure D-3, the pressure ratio of the turbojet engine
was taken as that value which gave the maximum thrust per unit air
flow. The compressor pressure ratio decreased with increased speed
and approached the value of 1.0 at 1400 miles per hour. At this
speed the turbojet approached a ram jet in operation; however, the
combustion-chamber temperature limit was maintained at 1540° F.

In the turbo-ram-jet engine by burning additional fuel in the
tail pipe to temperatures much higher than 1540° F, it was possible
to obtain considerably more thrust per unit engine weight and hence
the disposable load increased over that of the turbojet. In this
engine the compressor pressure ratio was likewise chosen to give
maximum thrust per unit of air flow and decreased with increase in
flight speed. The compressor pressure ratio approached a value of
1.0 at a flight speed of 1800 miles per hour and at this speed the
turbo-ram-jet engine approached a ram jet in operation.

In the case of the ram-jet engine, the range increased with
increase in flight speed and altitude for conditions investigated.
A number of flight speeds and altitudes are considered in the sec-
tion on the ram jet. In order to avoid confusion, only one flight
speed (2000 mph) is shown in figure D-3; the location of the points

at 30,000 and 50,000 feet are indicated. It is noted that the ram-jet engine gives better performance than the turbo-ram-jet or the turbojet engine and is approached by only the turbo-ram-jet engine when that engine approaches a ram jet in operation, that is, when the compressor pressure ratio approaches 1.0. The ram-jet engine has the disadvantage that at take-off the thrust is zero. The turbo-ram-jet engine has the advantage of providing good performance at high speeds and having appreciable thrust to assist in take-off.

The performance of the airplane equipped with the rocket engine varies with altitude and flight speed and is discussed in detail in the section on the rocket engine. One curve for a flight speed of 3000 miles per hour is shown in figure D-3 with the points for 100,000 and 50,000 feet indicated. Because of its lighter weight per unit thrust but higher specific propellant consumption, the rocket engine provides a slightly higher disposable load but considerably shorter range than the ram-jet engine. The rocket engine cannot compete with other engines on the basis of long-range aircraft operation, but it does have application for short-range operation where its simplicity and lightness of weight are important considerations.

In conclusion, it is again emphasized that these charts are not intended to be applied to the general selection of power plants for specific aircraft design problems, but are intended merely to provide perspective. For any specific aircraft design problem, a separate analysis is required with assumptions and conditions that accurately apply.

Flight Propulsion Research Laboratory,
    National Advisory Committee for Aeronautics,
        Cleveland, Ohio, April 21, 1947.

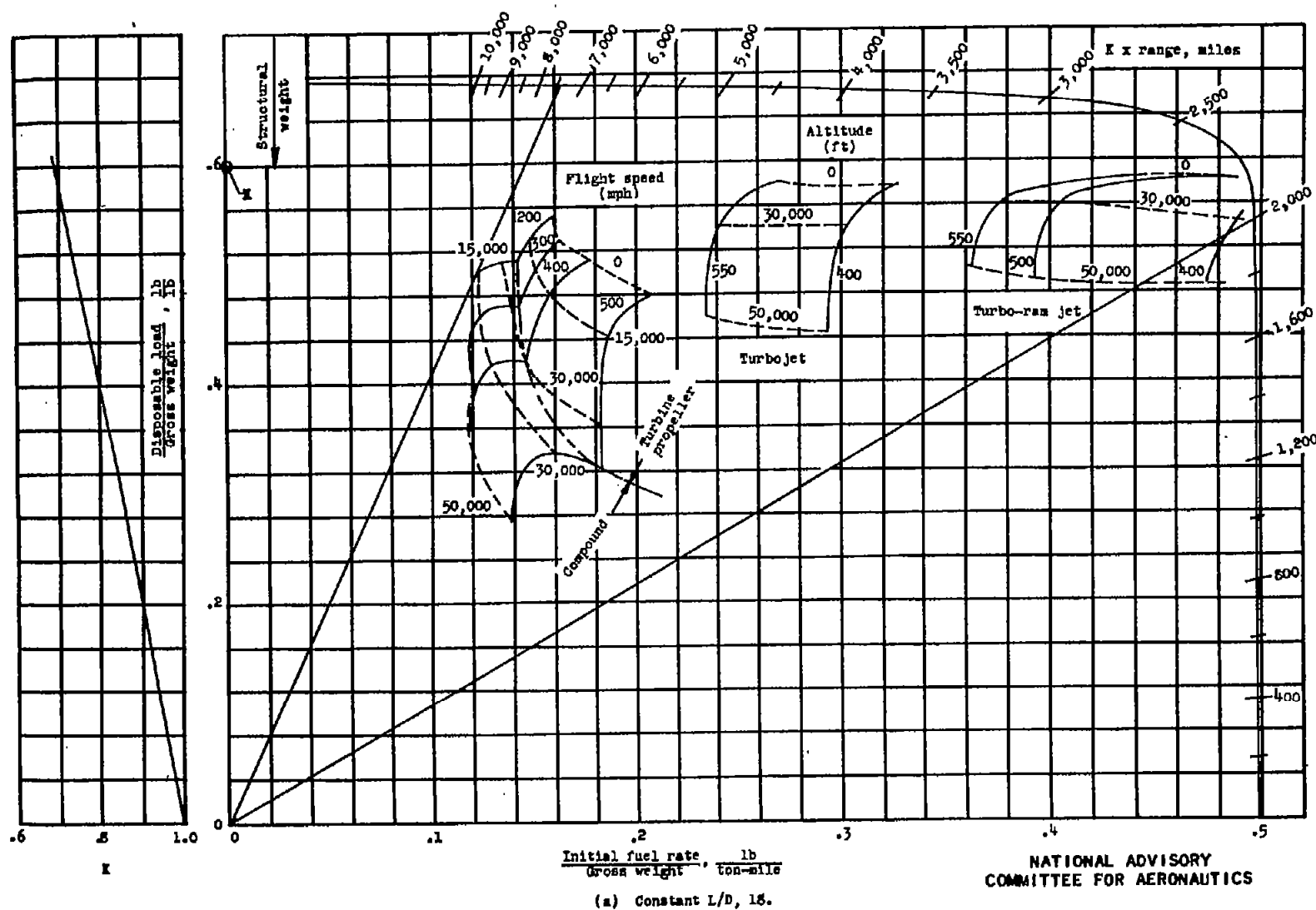(a) Constant L/D, 18.

Figure D-1. — Performance spectrum for subsonic flight.

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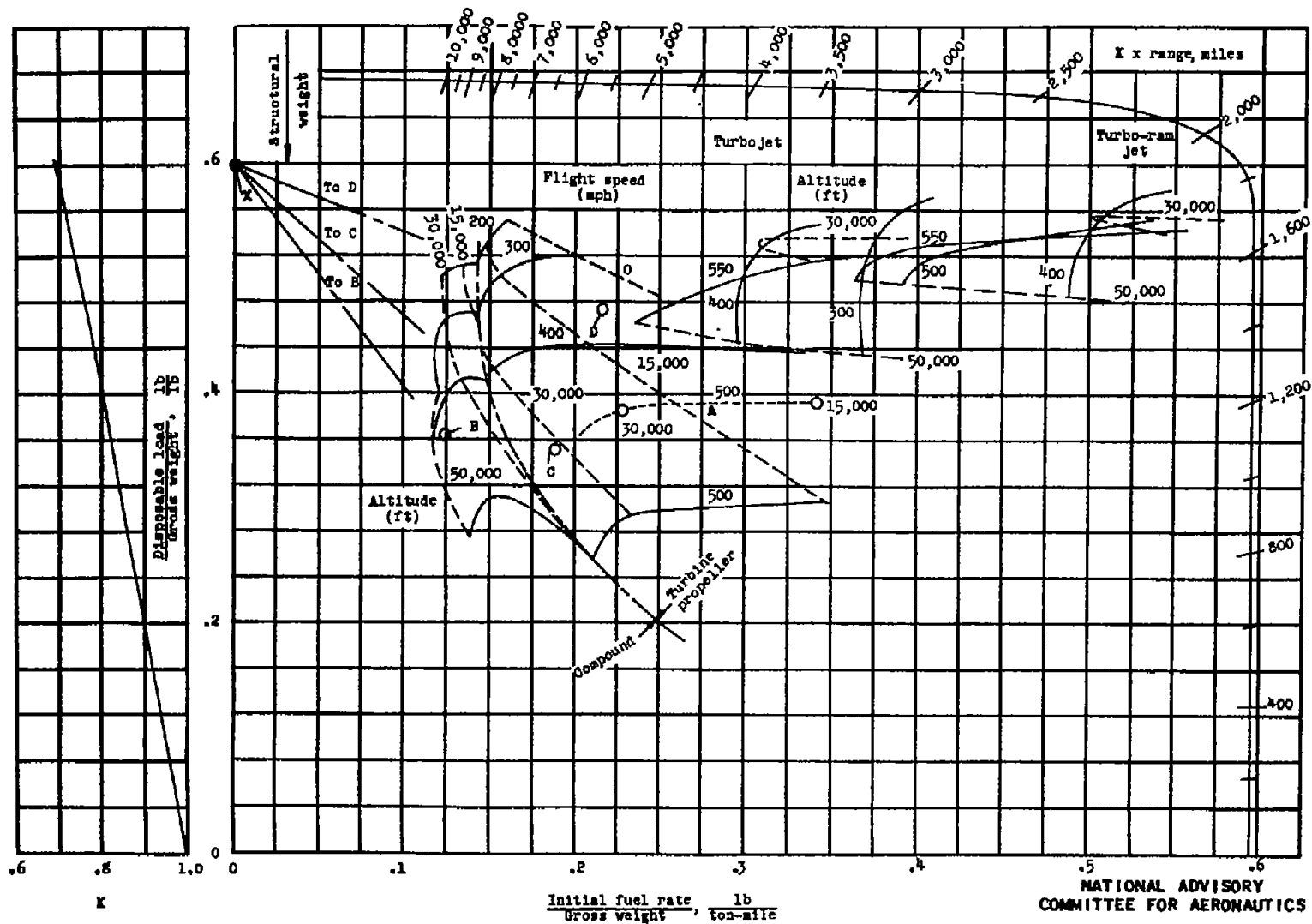
(b) Wing loading limited to 50 pounds per square foot.

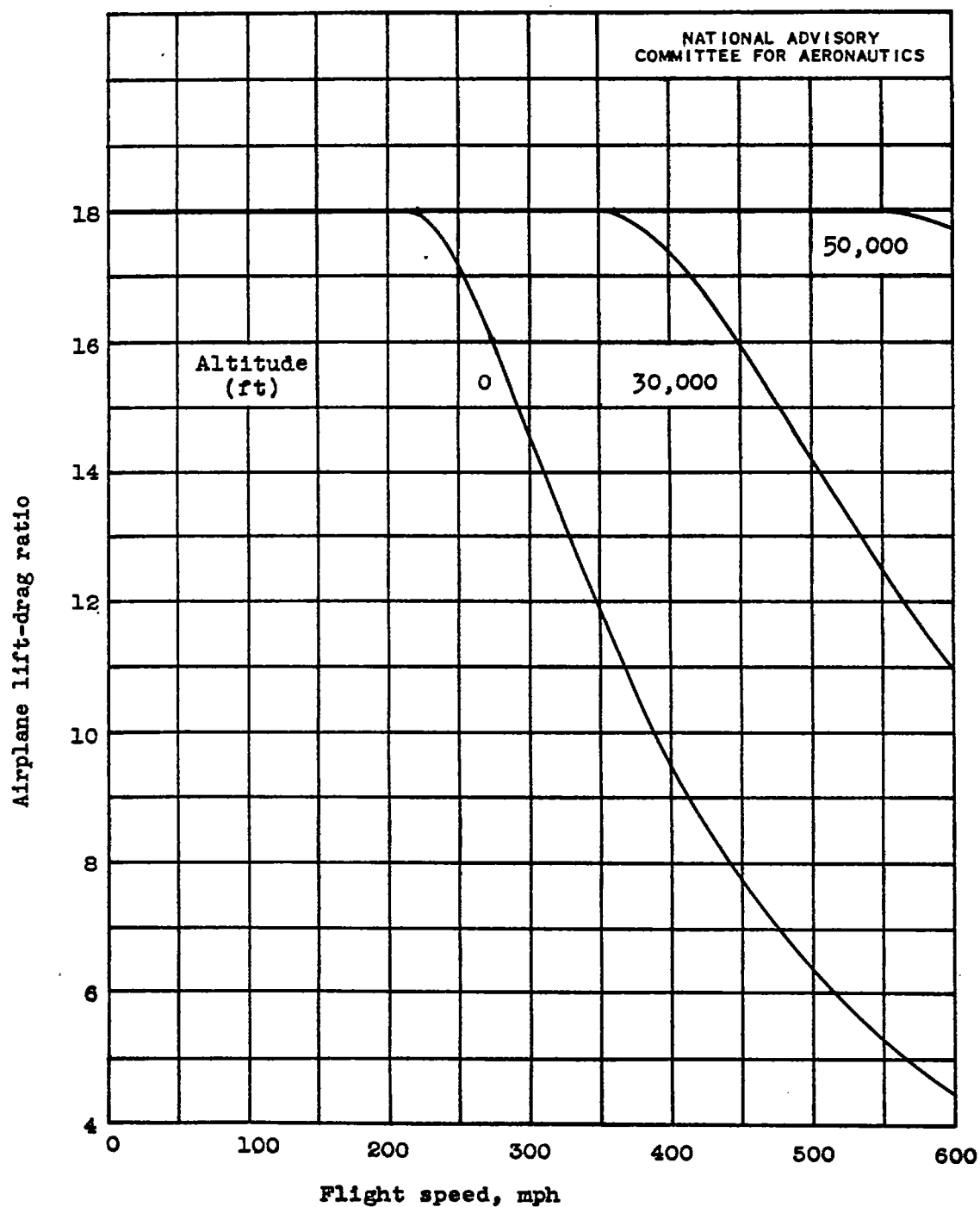Figure D-1. - Concluded. Performance spectrum for subsonic flight.

Figure D-2. - Variation of lift-drag ratio of airplane (minus nacelles) with flight speed and altitude for limiting wing loading of 80 pounds per square foot ($C_{D,0}$ = 0.019, $L/D_{max.}$ = 18).
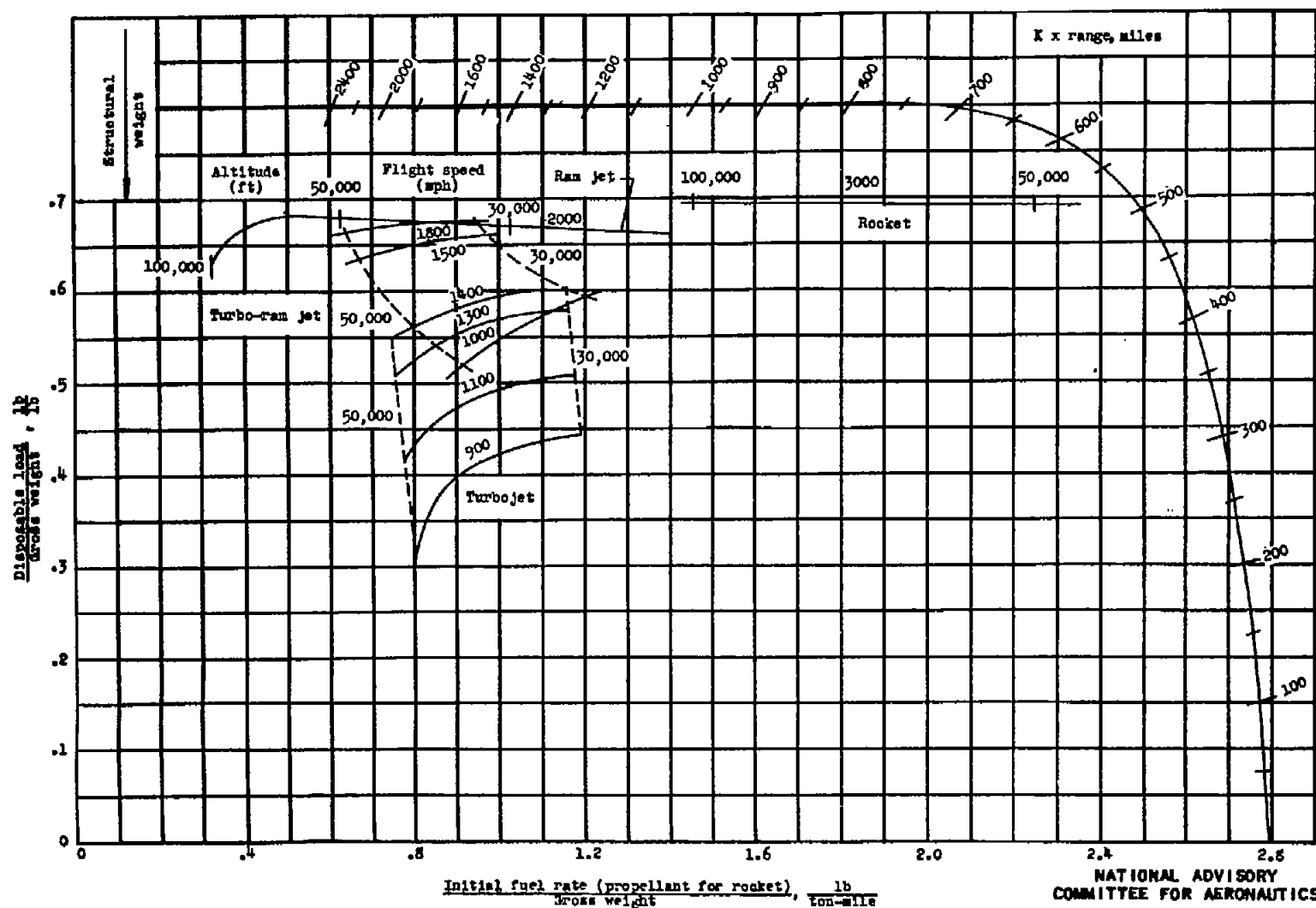
Figure D-3. — Performance spectrum for supersonic flight.

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APPENDIX - GENERAL ASSUMPTIONS

In order to insure a fair comparison of all the engine types considered, components used by more than one engine were assumed to have equal efficiencies on each of the engines. For example, for all the engines, the inlet diffuser was assumed to recover 90 percent of the dynamic pressure in the subsonic speed range; in the supersonic speed range, the ratio of the total pressure at the diffuser exit to the total free-stream total pressure P_d/P_0 was assumed to vary with Mach number M_0 in the following manner:

M_0	1.0	1.5	2.0	2.5	3.0	3.5
P_d/P_0	0.960	0.957	0.937	0.877	0.802	0.717

The nacelle drag in the subsonic speed range was based on maximum nacelle cross-sectional area and the drag coefficient C_D varied with Mach number M_0 in the following manner:

M_0	0.2	0.4	0.5	0.6	0.7
C_D	0.0556	0.0556	0.0560	0.0580	0.0655

In the supersonic speed range, the nacelle drag was evaluated by considering the drag as composed of two components, one due to skin friction and the second due to wave formation. The skin-friction drag coefficient was assumed to have a value of 0.003 based on wetted area and the wave drag was found as the product of the incompressible velocity head q, twice the maximum cross-sectional area minus the nacelle-inlet and the nacelle-exit area, and the wave-drag coefficient $C_{D,I}$, values of which are given in the following table:

M_0	1.0	1.2	1.5	2.0	2.5	3.0	3.5
$C_{D,I}$	0.10	0.129	0.106	0.086	0.074	0.064	0.054

For the propeller engines, the propeller efficiency η_p was assumed to vary with flight Mach number M_0 in the following manner:

M_0	0.2	0.4	0.6	0.7	0.8
η_p	0.85	0.85	0.85	0.82	0.70

The propeller weight was assumed to vary with engine shaft power, flight velocity, and altitude, where a sufficiently large

propeller was provided to attain the propeller efficiencies listed
in the preceding table. For the high-velocity, low-altitude cases,
these propellers are probably inadequate for take-off conditions.
For a shaft output of 2000 horsepower, the following propeller
weights were used:

Altitude (ft)	Flight speed (mph)				
	100	200	300	400	500
	Propeller weight (lb)				
0	1090	820	593	442	300
30,000	----	1810	1315	970	660
50,000	----	----	2740	2000	1360

For other shaft powers, the propeller weight W_p varied as
the 0.8 power of the shaft power

$$\frac{W_{p,1}}{W_{p,2}} = \left(\frac{hp_1}{hp_2}\right)^{0.8} \tag{A1}$$

For all turbine-type engines, a compressor efficiency of 85 per-
cent and a turbine efficiency of 90 percent were used. The effi-
ciencies were based on total temperatures and pressures. The steady-
flow combustion chambers used in turbine-type engines were assumed
to be 95 percent efficient, and the tail-pipe burner of the turbo-
ram-jet engine and the ram-jet combustion chamber were assumed to
be 90 percent efficient.

In determining the performance of aircraft using the various
engines, several assumptions had to be made concerning the aircraft
itself. The aircraft gross weight was considered to consist of
engine weight, fuel weight, structural weight, and pay load. The
engine weight was assumed to include engine accessories and propel-
ler. In the subsonic case, the structural weight, which included
nacelles and control-equipment weight but not fuel tank weight,
was assumed to be 40 percent of the gross airplane weight. The
tank weight was assumed to be 10 percent of the fuel weight. The
maximum lift-drag ratio of the subsonic airplane (minus nacelle
drag) was assumed to be 18. In cases where the wing loading was
limited to 80 pounds per square foot, the lift-drag ratio for any

operating condition was found assuming the profile-drag coefficient
of the aircraft less nacelles to be 0.019, while the maximum lift-
drag ratio for the aircraft remained at 18. . This is equivalent to
the assumption of an effective aspect ratio of 7.84. For these
assumptions, the lift-drag ratio can be maintained at 18 for flight
conditions with a value of q less than 117.0 pounds per square
foot with the wing loading below 80 pounds per square foot. For
higher values of q, the wing loading was held constant at
80 pounds per square foot and the lift-drag ratio was reduced
below 18 in accordance with the following equation:

$$\frac{1}{L/D} = 0.0002375 \, q + \frac{3.248}{q} \tag{A2}$$

With these assumptions as to the aircraft characteristics and
with a knowledge of engine performance, aircraft load-range
characteristics may be found. The disposable load per unit nacelle
frontal area is

$$\frac{W_d}{A} \quad \frac{W_g - W_e - W_s}{A} = \frac{0.6W_g - W_e}{A} \tag{A3}$$

where

A nacelle frontal area, sq ft

W_d total disposable load, lb

W_g gross weight of airplane, lb

W_e power-plant weight (including propeller), lb

W_s structural weight of airplane, $0.4 \, W_g$, lb

The aircraft gross load per unit nacelle frontal area is

$$\frac{W_g}{A} = \frac{(F - D_n) \, L/D}{A} \tag{A4}$$

where

F net thrust of engine, lb

D_n nacelle drag, lb

L/D lift-drag ratio of airplane without nacelles

From equations (A3) and (A4), the ratio of disposable to gross
load is

$$\frac{W_d}{W_g} = 0.6 - \frac{W_e/F}{\left(1 - \frac{D_n}{F}\right)(L/D)} \qquad (A5)$$

which determines the ordinate of the subsonic load-range curve.

The abscissa is found as follows:

$$\frac{w_f{}'}{A} = \frac{w_f}{AV_o} \qquad (A6)$$

where

$w_f{}'$ initial fuel rate, lb/mile

w_f fuel flow, lb/hr

V_o flight speed, mph

From equations (A4) and (A6), the abscissa of the load-range
curve is

$$\frac{w_f{}'}{W_g} = \frac{w_f/F}{V\left(1 - \frac{D_n}{F}\right)(L/D)} \qquad (A7)$$

When all of the disposable load is considered as fuel and tank load,
the range is a maximum and this indicated maximum range is determined
as the ratio of the ordinate to the abscissa on the load-range curve,
with a factor of 1.1 included to account for tank weight. The range
factor KR is obtained from equations (A5) and (A7) as

$$KR = \frac{W_d/W_g}{w_f{}'/W_g \; 1.1} \; \text{miles} \qquad (A8)$$

An exact evaluation of K, which is the ratio of the average
to initial fuel rate per mile per ton initial gross weight, involves
the complete flight plan as well as the engine and aircraft char-
acteristics. If it is assumed that the thrust power specific fuel
consumption of the engine and the drag-lift ratio of the airplane

vary linearly with the airplane gross weight, the general value
of K in terms of initial and final conditions can be derived:

$$\frac{1}{K} = \frac{\log_e \frac{f_0}{f_1}}{\frac{\Delta f}{f_0}\left(1 - \frac{f_0}{\Delta f}\frac{\Delta W}{W_0}\right)\left[1 - \frac{f_0}{\Delta f}\frac{\Delta(D/L)}{(D/L)_0}\right]} + \frac{\log_e \frac{W_0}{W_1}}{\frac{\Delta W}{W_0}\left(1 - \frac{W_0}{\Delta W}\frac{\Delta f}{f_0}\right)\left[1 - \frac{W_0}{\Delta W}\frac{\Delta(D/L)}{(D/L)_0}\right]}$$

$$+ \frac{\log_e \frac{(D/L)_0}{(D/L)_1}}{\frac{\Delta(D/L)}{(D/L)_0}\left[1 - \frac{(D/L)_0}{\Delta(D/L)}\frac{\Delta f}{f_0}\right]\left[1 - \frac{(D/L)_0}{\Delta(D/L)}\frac{\Delta W}{W_0}\right]} \tag{A9}$$

where

$(D/L)_0$ initial drag-lift ratio, lb/lb

$(D/L)_1$ final drag-lift ratio, lb/lb

$\Delta(D/L)$ $(D/L)_0 - (D/L)_1$, lb/lb

f_0 initial thrust power specific fuel consumption,
        lb fuel/thrust hp-hr

f_1 final thrust power specific fuel consumption,
        lb fuel/thrust hp-hr

Δf $f_0 - f_1$, lb fuel/thrust hp-hr

W_0 initial aircraft gross weight, lb

W_1 final aircraft gross weight, lb

ΔW fuel burned = $W_0 - W_1$, lb

If the drag-lift ratio and the thrust power specific fuel
consumption remain constant during the flight equation (A9)
reduces to

$$K = \frac{\frac{\Delta W}{W_0}}{-\log_e\left(1 - \frac{\Delta W}{W_0}\right)} \tag{A10}$$

which would also follow from the Breguet range equation.

If equivalent values used in the analysis are substituted in this equation, the simplified K as used on the load-range curves is found.

$$\Delta W = \frac{W_d}{1.1} \tag{A11}$$

where the disposable load is all fuel and tanks

$$W_0 = W_g \tag{A12}$$

$$K = \frac{\frac{1}{1.1}\frac{W_d}{W_g}}{-\log_e\left(1 - \frac{1}{1.1}\frac{W_d}{W_g}\right)} \tag{A13}$$

It is to be noted that ΔW is not equal to W_d. This is because the fuel tank weight was included in the disposable load; but inasmuch as the tanks were not considered expendable, ΔW was taken as equal to the fuel load only $\left(\frac{1}{1.1} W_d\right)$. The value of K computed from equation (A13) is plotted to the left of the ordinate W_d/W_g for all subsonic load-range figures.

For conditions where less than the maximum range is required, all of the disposable load is not used for fuel and tanks. In these cases, the ratio of fuel plus tank weight to initial gross weight is equal to $K \times range$ times the ratio of initial fuel rate per mile to the initial gross weight.

This value can be obtained graphically on the load-range plot by drawing a vertical line through the operating point and another straight line joining the origin to the desired value of $K \times range$. For this range, the vertical distance from the intersection of these two lines to the abscissa gives the desired value of fuel (plus tank) weight per unit initial airplane gross weight. The K curve previously described is entered at the ordinate of this intersection to determine the corresponding value of K.

In the supersonic case, the structural weight (less tank weight) was assumed to be 30 percent of the gross weight and the fuel-tank weight to be 10 percent of the fuel weight. The lift-drag ratio of the supersonic wing $(L/D)_w$ was assumed to be 7, and the fuselage drag was calculated in the same manner as the

supersonic nacelle drag. The fuselage size was determined by
making the fuselage large enough to hold the maximum amount of
fuel that could be carried at each operating condition. The
fuel was assumed to have a density of 50 pounds per cubic foot
for the turbine engines, 45 pounds per cubic foot for the ram
jet, and 62.4 pounds per cubic foot for the rocket. The fuse-
lage volume was found by assuming the fuselage to be a cylinder
with conical ends with an included angle of 20°, the over-all
fuselage fineness ratio being 12. For control volume in all
supersonic cases, 2 cubic feet per ton of aircraft was allowed
in the fuselage.

With these assumptions as to the aircraft characteristics
and with known engine performance, the aircraft load-range
characteristics can be found. The disposable load of the air-
craft is

$$W_d = W_g - W_e - W_s = 0.7 \ W_g - W_e \qquad (A14)$$

The gross weight of the aircraft is

$$W_g = (L/D)_w \ (F - D_n - D_f) = \frac{W_d + W_e}{0.7} \qquad (A15)$$

where D_f is a drag due to the fuselage and is found by the fol-
lowing equation:

$$D_f = q_o \left(\frac{W_d}{\rho_f} + \frac{W_d + W_e}{700}\right)^{2/3} \left(0.4528 \ C_{D,I} + 8.34 \ C_{D,F}\right) \qquad (A16)$$

where

q_o incompressible dynamic pressure $(1/2 \ \rho \ V_o^2)$, lb/sq ft

ρ_f fuel density, lb/cu ft

$C_{D,I}$ wave-drag coefficient

$C_{D,F}$ skin-friction drag coefficient, 0.003

The term $\dfrac{W_d}{\rho_f}$ in equation (A16) is the volume of fuel required
and the term $\dfrac{W_d + W_e}{700}$ is the control volume allowed in each case.

By combining equations (A15) and (A16), an equation involving
only the engine weight and the disposable load can be found from
which it is possible to determine the disposable load by trial-and-
error solution

$$(L/D)_w \left\{ F - D_n - \left[q_o \left(\frac{W_d}{\rho_f} + \frac{W_d + W_e}{700} \right)^{2/3} (0.4528 \; C_{D,I} + 8.34 \; C_{D,F}) \right] \right\}$$

$$= \frac{W_d + W_e}{0.7} \qquad\qquad (A17)$$

With disposable load known, the gross weight can be calculated and
the fuel rate per ton-mile then determined as in the subsonic case.

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